Evolutionary Strategy for the Use of Nuclear Electric Propulsion in Planetary Exploration

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Abstract. Given the recent advancements in power generation, waste heat rejection systems and electric propulsion, a reassessment of the benefits of Nuclear Electric Propulsion (NEP) is provided. Six different planetary missions are evaluated: a Pluto rendezvous, a Europa rendezvous, a Titan/Saturn sample return, a Europa sample return, a fast Mars piloted mission and a fast Neptune piloted mission. These various missions are evaluated against three major power levels which constitute an evolutionary path for the technology: 1) a 100-kWe relatively near term power system with ion engines, 2) a 1-50-MWe power system with either ion engines or magnetoplasmadynamic thrusters (MPDTs), 3) a 100-MWe power system with MPD thrusters. Detailed NEP vehicle mass breakdowns are established and combined with parametric low thrust trajectories. Delivered payload mass as function of trip times for each mission is provided. The analysis shows that NEP has applications over a large spectrum of missions. NEP is especially applicable for short trip time, large launch masses and high-energy missions.

INTRODUCTION

New power and propulsion technology efforts such as the DS-1 ion propulsion system demonstration, the recent funding of solar sail technology after many years of neglect and renewed interest in space nuclear power sources call for a reassessment of the mission types for which each technology is the most applicable. This paper focuses on Nuclear Electric Propulsion (NEP) as a means to transport large payloads to targets in the solar system which are energetically difficult to reach.

The main objective of this study is to assess as thoroughly as possible the benefits of NEP. A large emphasis has been placed in defining the NEP vehicle configuration and corresponding subsystem elements in order to produce an estimate of the vehicle's payload delivery capability which is as credible as possible. Therefore a few design points were studied, and reasonable interpolations were made around those design points. We believe the results provide a strong basis for comparing with other technologies (provided that the other technologies are also defined in such a level of detail).

The study is divided in three parts. The first part describes a 100-kW_e class NEP vehicle and evaluates its delivered payload capabilities for near term robotic planetary missions. The second part describes 1-MW_e to several tens of MWe class vehicles with two different propulsion systems and evaluates their performance and relative benefits. These systems are evaluated for both robotic and piloted missions. The third part describes the assumptions and results for a 100-MW_e vehicle for an outer planet piloted mission. All detailed mass breakdowns and graphs for the seven missions are provided in a special section as the end of the paper.

FIRST STEPS: ROBOTIC EXPLORATION OF THE SOLAR SYSTEM (100-KW CLASS POWER SYSTEM)

System Design and Vehicle Configuration

The 100-kW_e class propulsion system assumed here is derived from a Kuiper Belt Object Rendezvous study (JPL Team X, 1999) that involved several NASA centers and DOE Laboratories. The overall NEP vehicle configuration is based on the use of a SAFE-300 nuclear reactor and of an ion propulsion system. All subsystem masses assume the vehicle configuration shown in Figure 1. In this configuration, a long boom separates the power and propulsion systems from the other subsystems **of** the spacecraft. The boom also serves as a structural attachment for the deployable radiators. Every element of the vehicle other than the reactor is located in the reactor shield's shadow, which covers a 10° x **30**" solid angle. The power conversion system, propulsion system fuel **tanks,** feed system, power processing and thrusters are mounted next to the shield. The very large deployed radiators (about 130 m^2) are unfolded along each side **of** the main boom. In stowed configuration, the spacecraft fits within a Delta **IV** launch fairing (5-m diameter by about 14 m long).

Figure 1. Kuiper Belt Object Rendezvous Mission (JPL Team X, 1999), vehicle configuration.

Figure 2 shows a system level block diagram of the NEP design, which is common to the 100-kW_e and higher power cases.

Power System

The 100-kW_e system is the result of a detailed trade study in which a variety of reactor concepts and conversion systems were evaluated. The baseline system has a NASA/Marshall Space Flight Center (MSFC) SAFE-300 UO_2 fueled, heat-pipe-cooled reactor that has been extensively analyzed by NASA/MSFC and Los Alamos National Laboratories. The 100-kW_e system produces 102.4 kW_e electric and approximately 320 kW_{th} thermal. A system schematic is shown in Figure 3. Thermal power is transferred from the heat pipes in a molybdenum primary heat exchanger into the helium-xenon working fluid. The hot, high-pressure fluid is routed to a turbine where energy is extracted.

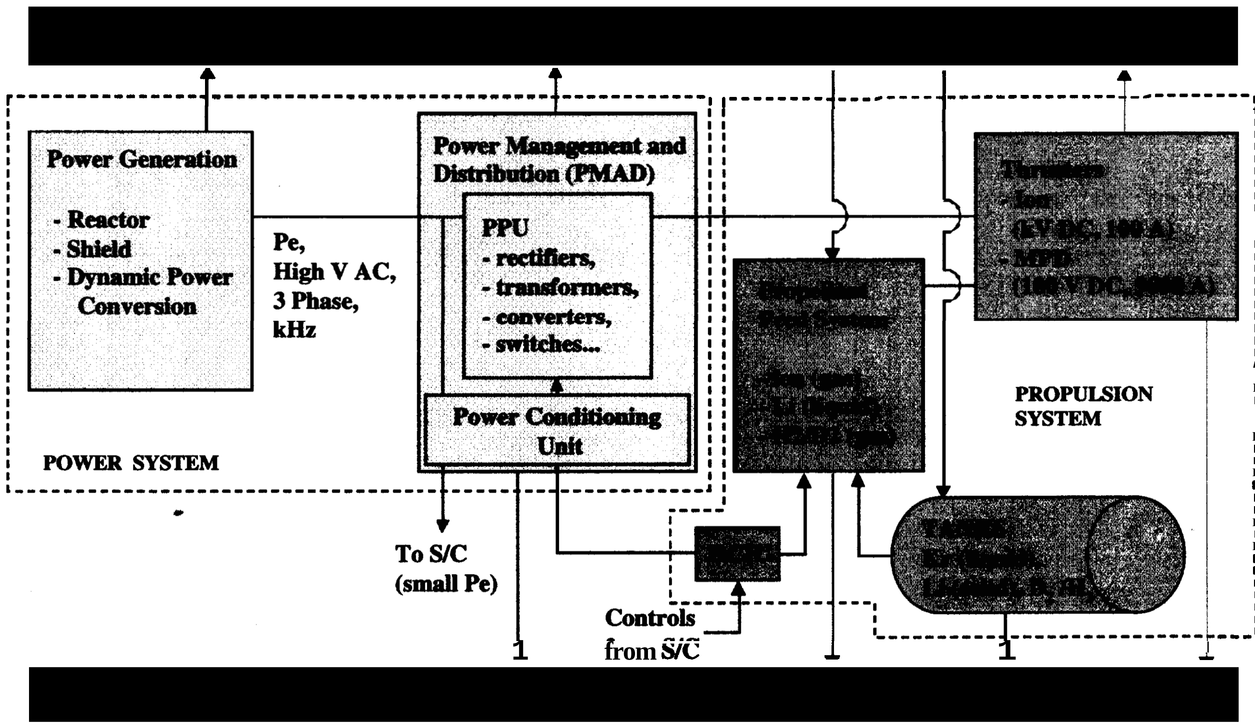


Figure 2. NEP system level block diagram.

Excess thermal energy is released into a rotary regenerator that heats up the working fluid heading toward the primary heat exchanger. The cooled working fluid is directed toward a capillary pumped loop (CPL) evaporator unit, where low-grade thermal energy is exchanged with the CPL radiator. The low temperature, low pressure working fluid is routed to the compressor. The compressor increases the pressure of the working fluid, and routes it to the rotary regenerator where it is heated by the regenerator prior to entering the reactor once again.

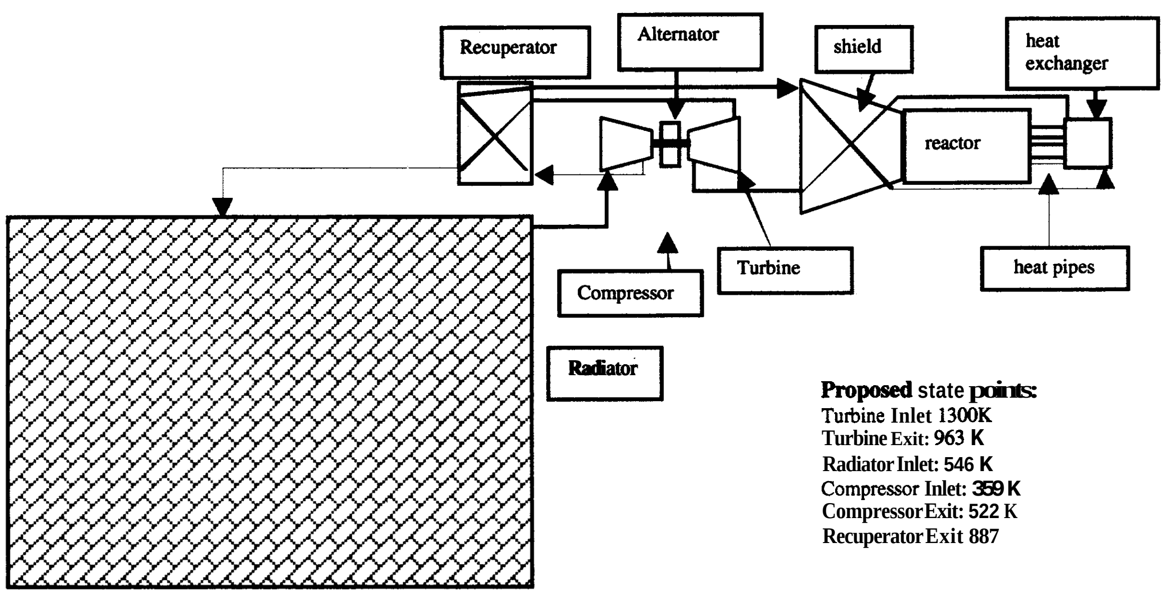


Figure 3. 100-kW$_e$ Power Generation System (Brayton cycle)

The major features of this system have been extensively analyzed and conceptually designed. Sandia National Laboratories, Marshall Space Flight Center and Los Alamos National Laboratory have evaluated many different reactor concepts. Depending upon mission application, some reactor systems are lighter in weight than others; however, the team has determined that for a given thermal power level, reactor mass estimates may vary from roughly 350 kg to 1100 kg, or about a factor of three. In selecting the heat-pipe-cooled concept, particularly attractive for its lighter weight at lower thermal power levels, we believe that a good match between required thermal power and upper temperature limits has been achieved. Lower mass systems, such **as** SNAP-8 derived models, cannot achieve the necessary temperature to allow high efficiency Brayton systems; by contrast, full flow direct gas cooled concepts are on the heavier end of the mass scale for this thermal power level.

The nuclear reactor is launched cold and is not started until the vehicle has reached an **Earth** escape orbit.

The Brayton cycle power conversion system has been analyzed in great detail by Lenard and Allison Advanced Development Center (Hansen, 2000). The temperature range and power levels have been scaled from the Allison Ma-1A Battle Tank turbine, although an independent analysis for the helium – xenon working fluid was performed. The turbine was designed for 360 kW_e **output,** but the power was reduced while the mass remained constant, so a conservative estimate was generated. The alternator was scaled based on mass estimates from Ashman Technologies (Hansen, 2000) who develops advanced alternators for various defense applications. Depending upon system rotational speed, alternator mass is almost insignificant, approaching 0.01 kg/kW_e at 90,000 **rpm** shaft speed. The analysis performed by Allison indicates that the turbo-machinery and alternator will easily meet the contractual requirements of **30,000** continuous hours of operation. The blade stresses are very low and consistent with long life. The major item of concern is the use of very lightweight rotating recuperators. These items are presently maintained in terrestrial airborne applications. Further study is required to ascertain whether or not they can be made for extremely long life in a non maintained system.

The next most massive component is the Capillary Pumped Loop radiator (Dynatherm, 2000). While a detailed mass breakdown was not available, Dynatherm had conducted several studies for water-CPL systems in this temperature range. The radiated power level was such that employing aluminum or stainless steel panels and fins, and aluminum or stainless steel tubing in the panel sections resulted in a net mass of 2.3 $\mathbf{kg/kW_{th}}$ (radiated power) including evaporator units and structure. Given the power radiated, a mass based on this specific mass was calculated.

The Power Processing Unit (PPU) design was provided by Luis Pinero from NASA Glenn Research Center (GRC) for the Kuiper Belt Object Study (JPL Team **X,** 1999). Figure 4 shows the PPU architecture. The mass and complexity were greatly reduced by tuning the output voltage of the turbo-alternator to a value close to the thruster inputs demand (direct-drive architecture). The design of the turbo-alternator is flexible enough to allow for this tuning. The efficiency **of** this PPU was estimated at 0.94. Each PPU processes 25 kW of power, so under normal operations four PPUs are operated simultaneously. The PPUs are designed to have a lifetime much greater than the thrusters, and thus only one spare PPU was included.

Propulsion System

A simplified block diagram of the ion propulsion system, which was also derived from the Kuiper Belt Object Rendezvous study, is provided in Figure 2. The ion propulsion system (IPS) is composed of 60-cm diameter ion engines that *can* process 25-kW of electric power and use krypton rather than xenon **as** propellant. The thruster has an estimated efficiency of 0.67 at a specific impulse (Isp) **of** 5000 s, 0.77 at an Isp of 10000 s, and 0.77 at **15000** s. At 25 kW and 9500 s of specific impulse, the thruster beam voltage is 5300 V, the beam current is 2 A, the accelerator grid voltage is 935 V, and the discharge propellant efficiency is **0.88.** The propellant throughput capability of each engine was estimated to be 500 kg, by scaling the capability of **an** existing, flight qualified 2.3 kW engine. The thrusters are powered by the PPUs, which convert the power from the turbo-alternator to the voltages and currents required by the engine. The feed system and PPU are controlled by the Digital Control Interface Unit (DCIU), which accepts and executes high-level commands from the spacecraft computer and provides propulsion subsystem telemetry to the spacecraft data system.

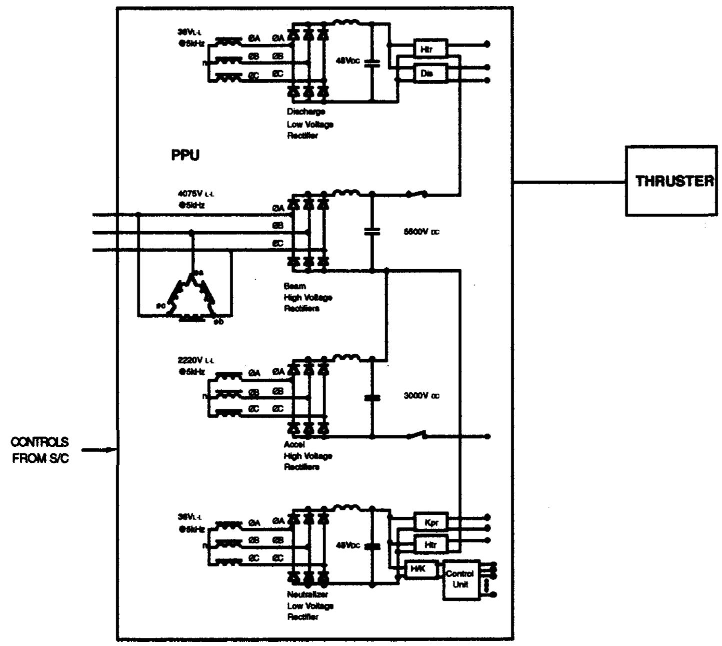


Figure **4.** 25-kW PPU direct-drive design. (Provided **by Luis** Pinero, **NASA GRC.)**

Although the design of the ion engines would be new, it is based on the experience and technology of the 30-cm, 2.3-kW NSTAR engine that flew on the New Millennium Deep Space-1 mission (DS-1) in 1998. Very little development of high-power ion thrusters has occurred and new technologies will be required, but none appear to have major feasibility issues. NASA is currently developing a 75-cm ion engine with capabilities similar to those assumed in this study (Patterson, 2000).

The tankage fraction was calculated assuming two cylindrical composite tanks. Those tanks have a propellant storage efficiency (including tank shell and insulation) of about **0.5%** (Lewis, 2000) for krypton, when stored **as** a liquid **at** 120 K and 1.4 Bar (20 psia). To keep the krypton stored below 120 K, a passive cooling system that uses a Sun shade and a V-Grove isolation system was designed. The Sun shade uses Silver-FEP Teflon on the Sun side, and is sharply angled. The side of the Sun shade facing the tank has a 20 layer Multi-Layer Insulation (MLI) blanket. Also between the tanks and the **Sun** shade is a double V-Grove radiation shield. With this design at 1 AU, the temperature of the tank is kept well below 120 K. The mass of this system is dependant on the total propellant mass and tank size. For 10,000 kg of Krypton, the total mass is 75 kg, including the **Sun** shade, V-Grove isolation, MLI for tanks and support structure. This mass was scaled to accommodate different propellant loads. Feed system design is similar to the Comet Nucleus Sample Return (Brophy, 2000) feed system. It is an improved version of the DS-1 feed system (no plenum **tanks)** and involves some new components (variable regulator, flow control devices) that should be available by 2010.

System Masses and Efficiencies

Table 1 summarizes the system masses, efficiencies and assumptions for the 100-kW_e class missions assuming 2000 kg of deterministic **AV** propellant'mass. Since each mission trajectory we considered has a different propellant mass requirement and thus a different number of engines, the system masses have been adjusted for each trajectory case. The mass list includes all the subsystems involved in the NEP vehicle but not the spacecraft bus-related subsystems such **as** telecommunications and avionics.

One spare ion engine for every four operating ion engines and one spare PPU and digital control interface unit (DCIU) are included for single-fault tolerance.

The structures/cabling masses are not based on a specific design but are a percentage of the subsystems to which the structures apply (typically 26% of the propulsion system and 16% of the power system for structures). These percentages) are based on historical data and are consistent with the design guidelines of the JPL integrated project design center (Team X).

The 30-m deployable boom is a similar design to that of the AEC-Able FASTmast, which was .designed for the International Space Station. FASTmast **is** a 35-m mast system that supports the Space Station large blanket solar arrays. The total mass of this mast including canister, deployment system and mast is about 300 kg. The mass used in this study was **80%** of the FASTmast value, under the assumption that this technology would improve within the next decade.

Pitch and yaw control of the vehicle is assumed to be performed with the ion engines. Roll will be achieved by a separate system (cold gas). Also, a two-degree-of-freedom gimbaled momentum wheel will be used to cancel momentum from the turbine in the power conversion system.

This mass breakdown varies with the propellant mass and number **of** engines. When necessary, the power system was scaled **as** the square root of the ratio of the desired power to 100 kW_e. This scaling was done close to the 100 kW_e point design. Figure **5** shows a percentage of the 100-kW_e vehicle dry mass breakdown by subsystem.

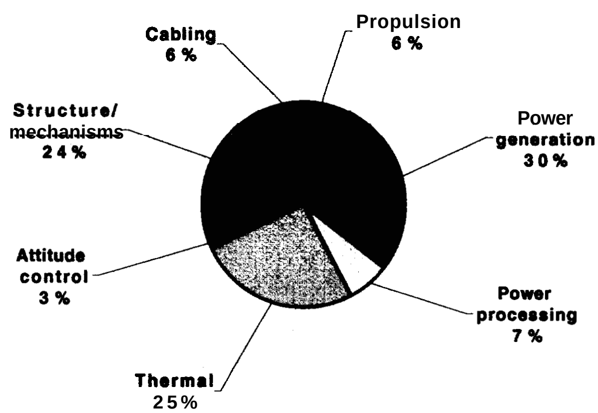


Figure 5. 100-kW_e system *dry* mass breakdown percentages per subsystem (for 2000 kg of Krypton). Total *dry* mass: 3184 kg, including 30% contingency.

Mission Results

The 100-kW_e class NEP system was evaluated for several mission types: a Pluto orbiter, a TitadSaturn sample return, and a Europa Orbiter mission. All trajectories were computed by Carl Sauer at JPL. They were run parametrically **as** a function of the ratio **of** initial power in kW_e to the initial mass in kg. They assumed the efficiency profile described above of the 25-kW ion thrusters and PPU, a tankage fraction **of** 10% and a duty factor of 100%. Each trajectory was optimized for specific impulse (Isp) although the maximum Isp was constrained to 16000 s to be consistent with the assumed thruster technology. Although parametric results were originally provided, the actual calculations of the net delivered mass assumed the launch from either a Delta IV M+ [5,4] (similar in performance to the Atlas V 521) or a Delta IV Heavy. With 10% derating and a C3 slightly above **0** km^2/s^2, the Delta IV M+ injected mass was assumed to be 4060 kg and the Delta IV Heavy injected mass 8325 kg.

The trajectory parametric results for these three missions are given in Figures 6, **8** and 10 **as** a finction of trip time and Po/Mo, where Po refers to the initial total electric power and **Mo** to the initial injected **mass.** Each trajectory started with a C3 slightly above 0 km^2/s^2. The net delivered mass (which is the total arrival mass at the target minus the dry NEP mass described in Table 1) is provided in Figures **7, 9** and 11.

The Pluto orbiter launch date was assumed to **be 2010.** The orbiter approaches Pluto with an excess velocity very close to 0 **km/s.** However, in the trajectory results, the vehicle does not get captured by the planet. An additional small AV would be required to capture and potentially circularize around Pluto. Here, the NEP AV is typically between 40-60 km/s. Propellant mass varies between **1000** and **4000** kg, leading to systems with ***5*** to **10** engines. As shown in Figure 7, a Delta IV M+ [5,4] **can** deliver to Pluto (rendezvous) a net mass slightly less than **200** kg in **11** years and about **350** kg in **12** years, while a Delta IV Heavy can deliver a net **mass** slightly above 600 kg in **9** years and about **2000** kg in **12** years.

The Titan/Saturn sample return mission leaves Earth in **2013,** and in about half the total trip time enters into a highly elliptical orbit around Saturn with apoapsis at Titan's altitude. There is a 5-month stay time in this elliptical orbit. The spiral in and out times to the elliptical orbit are very similar in both cases and are between **80** and **130** days depending on the trajectory. This mission assumes no mass drop at Titan, meaning that the NEP carrier brings back the same inert mass **as** it carried on the way there. This approach is conservative and was assumed since no Titan lander design was available at the time of writing. The return trajectory terminates with a fly-by of the Earth. The NEP AV for a Titan/Saturn Sample Return is typically between 50-65 **km/s.** Propellant mass varies between **2000** and **4000** kg, leading to systems with ***5*** to **10** engines. As shown in Figure **9,** a Delta IV Heavy can return to Earth a net mass a little 'bit less than **400 kg** in **10** years and about **1500** kg in **13** years.

The Europa orbiter mission assumes a launch in **2008.** The vehicle reaches Jupiter with a low excess velocity and the NEP system is used to capture to **a** circular orbit of Europa's radius. Spiral-in times to this orbit are between **180** and **330** days. The NEP AV is typically between 50-60 **km/s.** Propellant mass varies between **1700** and **3200** kg, leading to systems with ***5*** to **8** engines. As shown in Figure **11**, a Delta IV Heavy can deliver a net mass between **1200** and **2100** kg in **3.2** to **4.2** years.

Note that **as** Po/Mo increases, the optimized Isp increases (the optimized AV for a given trip time only varies slightly, therefore an increase in power leads to an increase in Isp) and so does the trajectory performance. Note also that the delivered mass increases significantly with trip time, therefore additional margin could be found if there is flexibility to trade trip time.

SECONDS STEPS: OUTER PLANET SAMPLE RETURNS AND HUMAN EXPLORATION OF NEIGHBORING PLANETS (1-50 MW CLASS POWER SYSTEM)

With **1-50** MWe of power, fast, high energy robotic and piloted missions can be accomplished. The vehicle and subsystem assumptions are summarized here and mission results **as** a function **of** flight time are provided.

System Design and Vehicle Configuration

For the **1-10-MW$_e$** class missions, two propulsion systems were studied. The first one uses the same size ion engines **as** the **100-kW$_e$** class, but operated at 60 kW per engine. Nine of these engines clustered together forms a 480-kW engine equivalent **(1** engine is redundant). The description of this system is provided below. For this system, the same vehicle configuration **as** the **100-kW$_e$** case was assumed. The other propulsion system considered uses lithium-fuelled Lorentz Force Accelerators (LFA's), also known **as** Magnetoplasmadynamic thrusters (MPDT's). To avoid contamination of "cold" surfaces by condensation of lithium fiom the engines, a slightly different vehicle configuration which isolates sensitive spacecraft surfaces fiom the engines was used, **as** shown in Figure **12.** A plume shield was also added to reduce lithium contamination. This configuration was inherited from a detailed **Mars** cargo vehicle study performed in the early **90s** (Frisbee, **1993).** Experimental measurements of lithium backflow (Kim, **1995)** and numerical simulations of lithium thruster plumes (Samanta-Roy, **1992)** indicate that a combination of physical separation and plume shields will prevent spacecraft contamination. Other configurations might be possible where the thrusters could be located close to the reactor system **(as** for the ion engines).

The NEP power system is based on the SP-100 reactor design with dynamic power conversion. **Both** vehicles have a boom of 50-m total length. Here again, this deployable boom is similar to the AEC-Able FASTmast with an assumed 30% mass reduction by 2016.

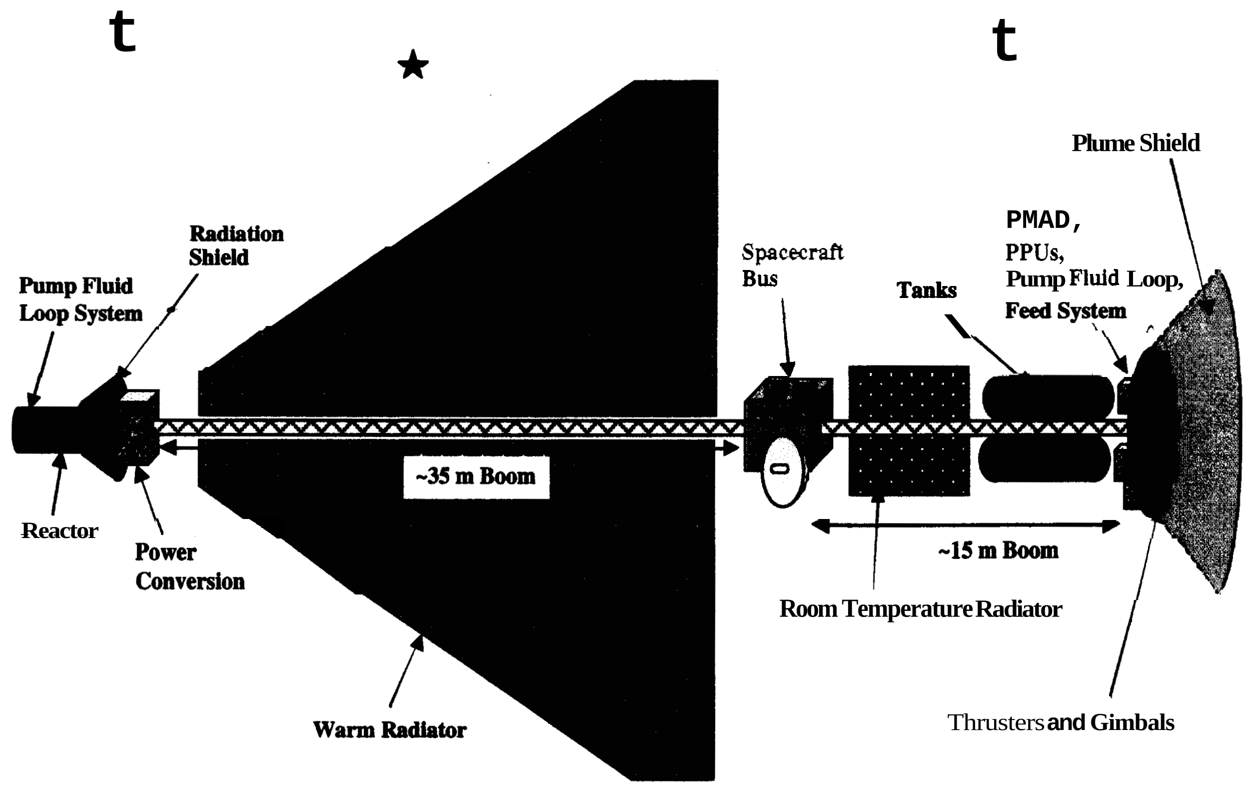


FIGURE 12. 1-MWe Vehicle configuration assumed for an MPDT based propulsion system.

Power system

A preliminary analysis and conceptual design **of** the major features of this system have been completed. Sandia National Laboratories, Marshall Space Flight Center and **Los** Alamos National Laboratory have evaluated many different reactor concepts. Depending upon mission application, some reactor systems that may be applicable for the 100- kW_e class are not viable for this power category. In selecting the direct gas-cooled concept, particularly attractive for its lighter weight at these thermal power levels, we believe that a good match between required thermal power and upper temperature limits has been achieved. The mass of shielding has increased to accommodate the increased reactor power level. We believe there **is also** the potential to scale this concept to at least the 10-MW_e class **of** power.

The Brayton cycle power conversion system has been extensively analyzed by Lenard and Allison Advanced Development Center (Hansen, 2000). The temperature range and power levels have been scaled from the Allison MA-1A Battle Tank turbine Auxiliary Power Unit, although an independent analysis for the helium–xenon working fluid was performed. The turbine was designed for 100 kW_e (shaft) output in the air-breathing mode, but the power was increased because of the higher density and energy content of the helium–xenon fluid. For the **1-MW_e** class mission the turbine speed was increased and the size was also increased over the previous 360-kW_e design point of the study. The alternator was scaled based on mass estimates from Ashman Technologies (Hansen, 2000) developed for the **360-kW_e** Allison study. The alternator design for the 1-MW_e case appears to be at the crossover point for incorporating high temperature superconducting stator windings. It does not appear feasible to increase alternator efficiency above about ***97%;*** this means that 30 kW_{th} will have to be rejected through the primary radiator. Detailed analyses of harness weight will also have to be traded to ascertain the utility of incorporating high temperature superconducting harnesses at this power level.

The next most massive component is the Capillary Pumped Loop radiator. The primary difference between the radiator for the **1-MW$_e$** concept and the **100-kW$_e$** class vehicle is the incorporation of lightweight, high thermal conductivity panels in the radiator. While water will still be the preferred fluid, it is desirable to use titanium tubing instead of stainless steel tubing. Since the turbo-machinery is larger, the efficiency increases, so less fractional thermal power must be radiated. Given the power radiated, a mass has been estimated using scaled Dynatherm values. Using the proposed changes, the mass should be reduced to about **0.55 kg/kW$_{th}$**. **As** in the 100-kW$_e$ case, Sandia National Laboratories has performed some analyses to ascertain the impact of deployed radiator panels in the non-shielded radiation field. Because the panels are so thin, they appear to cause minimal volume scattering effects. Consequently, the dose to the payload section appears to be small fiom the deployed radiator panels.

The PPU design for the ion engines was provided by Luis Pinero from **NASA** GRC. It assumes the engine cluster configuration described in the Propulsion section and one PPU per cluster (each PPU processes **500 kW** of power). Here again, the PPU is designed to be "direct-drive". Figure **13** shows the internal design of the PPU. The PPU total efficiency is about **98%.** Note that in this design, the assumed frequency is **5000 *Hz.***

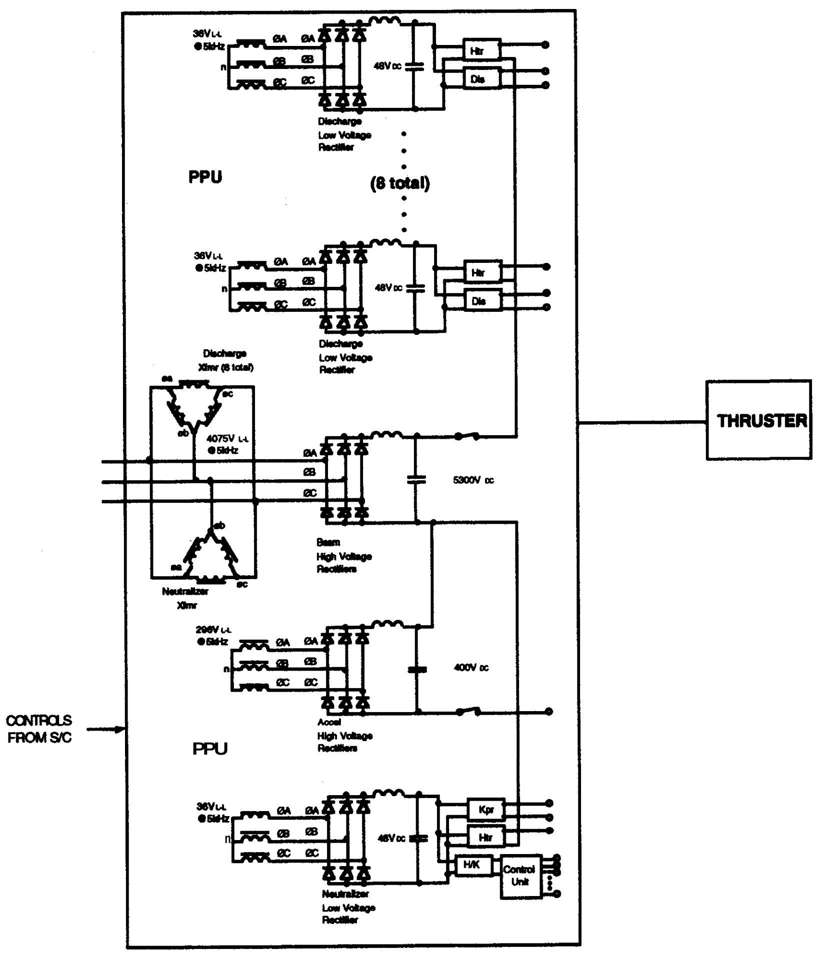


FIGURE 13. 500-kW$_e$ PPU conceptual design per ion engine cluster (Pinero, 2000)

The 1-MW PPU and cabling design for the LFA thrusters is a scaled version of the PPU design proposed in the early 90's for a Mars cargo mission (Krauthamer, 1995). This design assumes that the PPUs are located close to the thrusters such that it minimizes the cable mass between the turbo-alternator and the PPU. The transmission lines between the turbo-alternator and the Power Processing Module (PPM) are low-current, high-voltage (10000 **V,** 3-phase voltage of 5773 **V,** 5 kHz) lines. A transformer located in the vicinity of the PPUs and thrusters steps down the voltage to a 60-100 **V** level. The PPU then includes one Input switch array that drives three 500-kW controlled rectifiers and filters (including one spare) and two power conditioning units (one spare) for supplying the spacecraft bus with Housekeeping power. A controlled rectifier and filter output switch array feeds the thrusters with 60 **V** and **9000** A via large bus bars (current leads) that also serve **as** structural supports. Each thruster also includes a non-load break electromechanical switch. Line and electronics efficiencies are included in the detailed mass breakdown provided in the System Masses section.

Propulsion Systems

MPDT Propulsion System

Lorentz Force Accelerators (LFA's) are the only type of electric thruster with a demonstrated capability to process steady state power levels up to several MWe in a relatively compact device. In these engines a very high current is driven between coaxial electrodes through an alkali metal vapor (e.g., Li) or gaseous propellant. Lithium propellant yields very high engine efficiency because it has low frozen-flow losses. Because it has a very low first ionization potential and a high second ionization potential, very little power is expended in creating the plasma. The current interacts with a self-induced or externally-generated magnetic field to produce an electromagnetic body force on the gas. LFA's can operate efficiently at power levels fiom **150** kWe up to tens **of** MWe and are therefore ideally suited for a number of ambitious future in-space applications which require high power. Table **2** shows the projected MPDT performances.

Each 500-kW MPDT is able to process 4500 kg of lithium propellant. Lithium is stored **as** a solid and thus minimal tank strength is required. Waste heat from the thrusters is used to melt lithium at a temperature of 181°C. Two cylindrical tanks are used to store propellant on either side of the boom. The tanks and thruster waste heat transfer system have a tankage fraction of **2.8%** (Lewis, 2000). The feed system masses were estimated based on current MPDT feed system work done at Princeton University and JPL (Kodys, 2000).

Lithium-fed thrusters are also a stepping-stone to more advanced hydrogen-fueled thrusters, which can operate at many MWe per engine with Isp's 210,000 s and efficiencies ≥60 %. These engines could enable the use of multi-hundred megawatt nuclear electric vehicles, **as** discussed in the third part of this paper.

TABLE 2. Projected LFA thruster performance and lifetime.

Power **per** engine **(MW)**	0.5	1	2	5
Efficiency	0.55	0.65	0.65	0.65
Lifetime (hrs)	5000	5000	3000	2500
Throughput (kg)	4500	6300	16000	30000

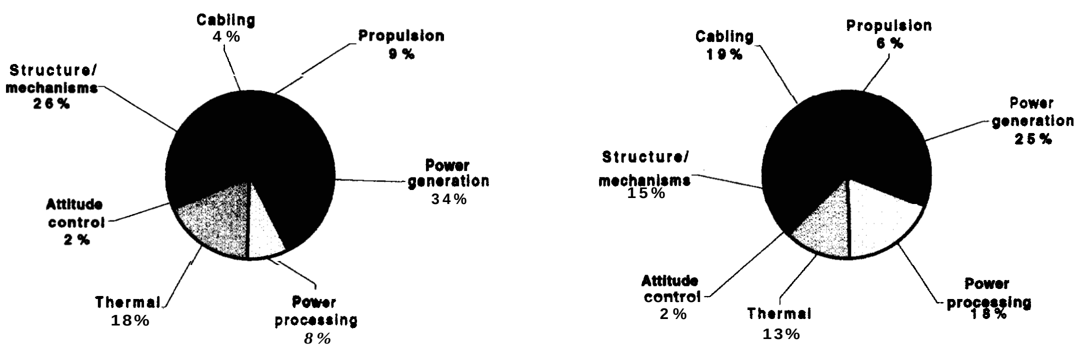
Ion Propulsion System

The 1-MW ion propulsion system is composed of two 480-kW ion engines. Each ion engine includes eight 60-cm, 60-kW ion sources plus one redundant source configured as a square 3 by 3 array. This approach is called a "segmented ion engine" in which multiple discrete ion sources are integrated together to form a single large ion engine with a large effective total grid area. A significant advantage of such a design is that the ground facilities and pumping requirements needed for testing are much relaxed (and feasible) relative to testing an equivalent single 480-kW ion engine. The 60-kW ion sources are essentially the same design **as** the previously described 25-kW engines. Each 60-kW source is assumed to have **an** efficiency between 0.65 and 0.75 at specific impulses between 5000 s and 16000 s. At 60-kW and 10000 s of specific impulse, the thruster beam voltage is about 3800 **V,** beam current for the segmented ion engine as a whole is 120 A, acceleration grid voltage is 400 V, and discharge current for each ion source is about 110 A. It is recognized that these characteristics are still preliminary since no test data are currently available. The engine lifetime has been estimated to be 240 kg per ion source, resulting in a total propellant throughput capability of about 1900 kg of krypton for the segmented engine. Each engine sits on a structure approximately 2.4 m wide and 3.0 m long. To fit within the launch vehicle fairing (about 4-5 m diameter), a deployment system has been added to deploy the two engines.

The krypton tankage assumptions along with the thermal control (Sun shade) for the ion propulsion option are the same as for the 100-kW_e case but scaled to the larger propellant loads for the 1-MW_e system.

System Masses and Efficiencies

Table 3 and 4 summarize the vehicle and subsystem mass breakdown. As can be seen from the tables for a 1-MW_e system, the vehicle mass with ion engines is less than that for the MPDTs. Although MPDTs have a higher power density than ion engines, the mass of cabling internal to the PPU or external to route the power largely offsets their advantage (MPDTs require around 100 V and 9000 A as inputs). An effort needs to be made in the power routing and processing to reduce the mass of cabling,, possibly with high temperature superconductors, to make it an equally attractive solution to ion engines at this power level.



Ion propulsion vehicle
Total dry mass: 7536 kg
(including 30% contingency)

MPDT propulsion vehicle
Total dry mass: 10011 kg
(including 30% contingency)

FIGURE 14. 1 MW system mass breakdown percentages per subsystem (for 3800 kg of propellant)

However, for missions that require higher power (> 2-3 MW_e), ion propulsion becomes less and less attractive due to the large number of engines required and their deployment, and their limited propellant throughput capability, which is lower than that of the MPDTs. Thus typically for human missions, MPDTs were a preferred option as a propulsion system.

The mass breakdown shown in Tables 3 and 4 varies with the propellant mass and number of engines. When necessary, the power system was scaled as the square root of the ratio of the desired power to 1 MW_e. This scaling was done in the range of 0.3-50 MW_e. Also, for some 1-MW_e trajectories, 3 to 4 ion engines were used as demanded by the propellant loads, but no consideration was given to the packaging within the launch vehicle fairing.

For robotic missions, a reactor shield mass of 700 kg was assumed whereas for piloted missions, a reactor shield mass of 1400 kg was used based on a Manned Mars Mission study (Rockwell International, 1989) to account for the lower radiation environment required by human presence.

Mission Results

All trajectories were calculated parametrically as a function of the ratio of initial power in kW_e to the initial mass in kg. They assumed the efficiency profile of the 60-kW ion propulsion system or of the MPDT propulsion system provided in the Propulsion section, a tankage fraction of 10% and a duty factor of 100%. Here again, each trajectory was optimized for specific impulse (Isp) although the maximum Isp was constrained to 16000 s for ion engines or to 8000 s for MPDTs to be consistent with thruster technology. All trajectories start with a C3 slightly above 0 km^2/s^2.

Robotic Missions

The 1-MWe system with ion engines was very well-suited for high-energy robotic missions. Here the TitadSaturn sample return mission was reassessed and a Europa sample return was considered. Power levels above 1 MW_e appeared to be more applicable to large payload deliveries, as for a piloted mission. All trajectories presented below assume an ion engine propulsion system.

As with the 100-kW_e case, the Titan/Saturn sample return mission leaves Earth in 2013, and in about half the total trip time enters into a highly elliptical orbit around Saturn with apoapsis at Titan's altitude. There is a 5-month stay time in this elliptical orbit. The spiral in and out times to the elliptical orbit are very similar in both cases and are between 30 and 60 days depending on the trajectory. This mission assumes no mass drop at Titan, meaning that the NEP carrier brings back the same inert mass as it carried on the way there. On the way back, the NEP vehicle performs an Earth fly-by. Figure 15 shows the trajectory results for the Saturn sample return for Po/Mo (initial power over initial mass) of 60, 80 and 100 W/kg as a function of round trip flight time. The trajectories were constrained by a specific impulse of 16000 s, and as Figure 15 shows, a higher Isp would have been desirable for an optimum trajectory. Figure 16 shows the net delivered mass (which is the total arrival mass at the target minus the dry NEP mass described in Table 3) for various initial masses. The NEP AV is between 30-60 km/s. Propellant mass varies between 2000 and 9000 kg, leading to systems with 1 to 5 engine clusters. As shown in Figure 16, a Delta IV Heavy can return to Earth a net mass slightly over 1000 kg in 10 years.

The Europa sample return mission assumes a launch in 2008. The vehicle reaches Jupiter with a low excess velocity and the NEP system is used to capture to a circular orbit of Europa's radius. Spiral in and out times to this orbit are between 100 and 180 days. No mass drop at Europa was assumed, which is conservative. The return trajectory terminates with a fly-by of the Earth. The NEP AV is typically between 40-55 km/s. Propellant mass varies between 2000 and 3000 kg, leading to a system with 2 engine clusters. As shown in Figure 18, a Delta IV Heavy can return to Earth a net mass of about 850 kg in 5.2 to 5.6 years.

Piloted Missions

One mission of particular interest is a piloted Mars mission. A possible architecture is to send from LEO a piloted vehicle that would deliver humans to an orbit around Mars and bring them back to Earth. Multi-MW NEP could be suitable for fast trip times of both a piloted vehicle and possibly cargo vehicles.

The trajectories were run by Carl Sauer at JPL. They assume a 2018 launch, start at 400-km LEO, and end in a 1-SOL elliptical orbit at Mars. Stay time at Mars is about 30 days. The specific impulse was kept constant at 5000 s (previous analysis showed that trajectories with an Isp of 8000 s were not feasible for trip times of 1 year). They also assume a constant thruster efficiency of 0.6 and PPU efficiency of 0.95. The return trajectory concludes with an Earth fly-by with an excess velocity of 9.4 km/s, which leads to atmospheric entry of a crew return vehicle of about 14 km/s. Round trip ΔVs are between 30 and 50 km/s. Propellant (lithium) masses were around 70 MT for the 10 MWe case, 160 MT for the 20 MWe case and 370 MT for the 50 MWe case.

The power levels considered varied between 10 and 50 MWe. Only 11 LFA thrusters (one redundant) with the capability to process between 1 and 5 MW each were assumed for each power level. The assumed propellant throughput capability per engine is shown in Table 2. Dry vehicle masses were computed using the mass list provided in Table 4 for MPDTs by scaling the power system as the square root of the ratio of the power needed to 1

MW. Dry vehicle masses were about 40.1 MT for the 10-MW_e case, 59.1 MT for the **20-MW_e** case, and 99.3 MT for the **50-MW_e** case.

Figure 19 shows the initial LEO mass **as** a function of trip time and Figure 20 shows the net returned mass at Earth fly-by. The result of this study is that a **20-MW_e** NEP vehicle could carry a 41 MT habitat to Mars in about 1 year. Its total initial mass in LEO (IMLEO) is about 236 MT.

THIRD STEPS: HUMAN EXPLORATION OF OUTER PLANETS AND THEIR MOONS (100 MW CLASS POWER SYSTEM)

System Design and Vehicle Configuration

For the 100-MW_e power levels, no attempt was made to define a vehicle configuration or to develop a detailed mass breakdown. Rather a set of requirements is provided in order to **perform** a piloted outer planet mission.

The power system could potentially be based on a Fissioning Plasma Core Reactor with Magnetohydrodynamic power conversion system (Knight and Anghaie, 2000). This power system could provide a power specific mass (with reactor, shield, radiators, structure, pumps and generator) **as** low **as** 0.5 **kg/kW_e** at 100 MW_e.

The propulsion system would be based on the used of hydrogen or deuterium MPD thrusters. These engines are predicted to be 24% efficient at an Isp of 5000 **s**, 47% efficient at 10000 s and 65% efficient at 15000 s (Choueiri and Ziemer, 1999).

Results for a Piloted Triton/Neptune Mission

The Triton/Neptune piloted mission leaves Earth in the 2020 time frame from a 1000 km altitude LEO. In about half the total trip time it enters into a highly elliptical orbit around Neptune with apoapsis at Triton's altitude. There is **a** 5-month stay time in this elliptical orbit. The spiral in and out times to the elliptical orbit are very similar in both cases and are between 3 and **50** days depending on the trajectory. This mission assumes no mass drop at Triton. The return trajectory assumes an Earth spiral-in back to a 1**000-km** LEO. The NEP **AV** is between 130-170 km/s. Propellant mass varies between **300** and 1400 MT.

Trajectories run by Carl Sauer at JPL assumed the D_2 MPDT efficiency profile and a 94% efficient PPU. Net returned mass results are provided in Figure 21 for a Neptune piloted mission **as** a function of trip time and total vehicle specific mass (excluding tank **mass).** IMLEO masses varied between 500 and 2000 MT. **A** tankage fraction of 16% was assumed to calculate the net returned mass.

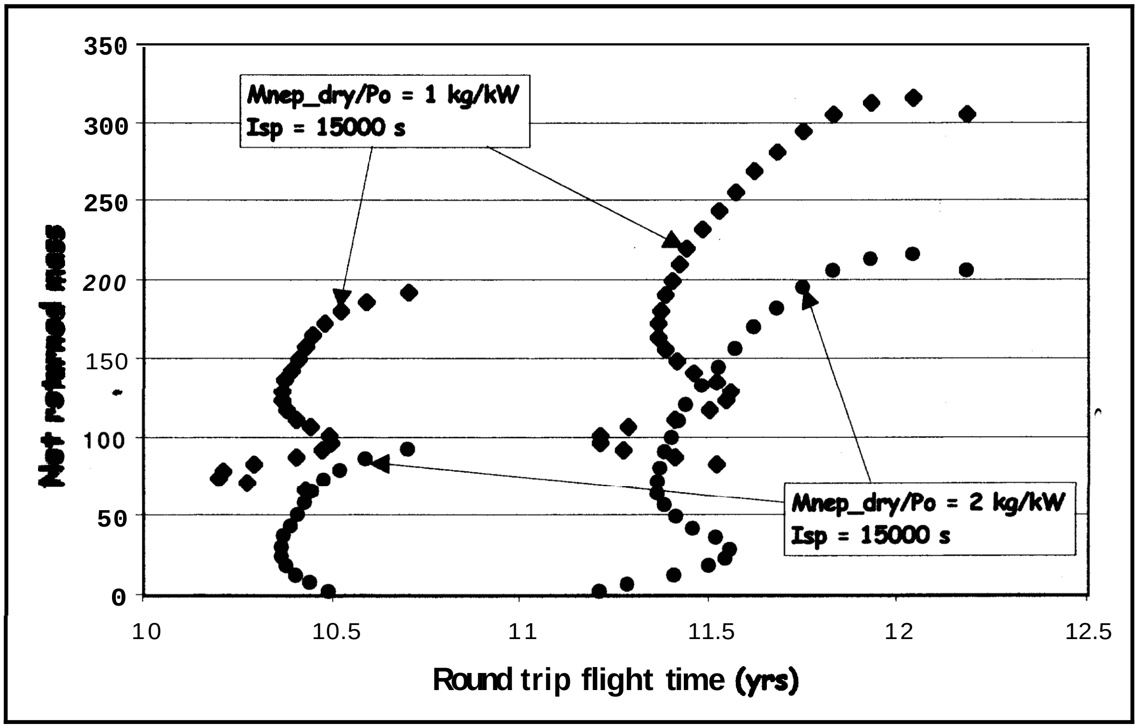


FIGURE 21. Neptune piloted **mission results for a** 100MWe vehicle.

CONCLUSIONS

This *study* reviews the potential applications of a range of NEP power and propulsion systems. It is found that 100-kW_e power and ion propulsion systems are applicable for a 9-12 year Pluto rendezvous, a 10-13 year TitadSaturn sample return and a **3-4** year Europa orbiter mission. Net delivered masses varied between **500** and 2000 kg. **As** the total *dry* **mass** of the 100-kW_e NEP is quite large (around 3200 kg), the benefit of NEP only shows for a Delta IV Heavy (or equivalent) launch vehicle. All robotic mission trajectories started from a slightly positive **C3.**

1-MW_e class NEP vehicles did not improve these robotic mission trip times for the same delivered payload (assuming the same launch mass). The TitadSaturn sample return trip times were 10-12 years on the Delta **IV** Heavy. This conclusion might be **an** artifact of the constraint in Isp to 16000 s. **A** higher Isp (up to 30,000-40,000 s would probably make the 1-MW_e vehicle look more promising (in terms of delivered mass) than the 100-kW_e vehicle. However, a 500-1000-kW_e class vehicle enabled a Europa sample return in ***5-6*** years.

Piloted missions, such **as** a Mars piloted mission in 1 year or a Neptune/Triton piloted mission in 11 years, required power systems from 10 to 100 MW_e since the mass of the transit habitats are large (several tens of MT). Here again, higher Isp would enhance mission results. For most high energy trajectories **an** Isp of 20000 to **40000** s would be more appropriate (for the same trip times). The piloted missions started fiom low Earth orbit and assumed an Earth fly-by for the Mars case and a return to LEO for the Neptune case.

Detailed **mass** estimates of the **100-kW_e, 1-MW_e** (with ion engines or with **MPD** thrusters), and **10-50-MW_e** vehicles are provided. The NEP vehicle specific mass as function of power level is summarized in Table 10. $M_{NEP(veh-tk)}$ represents the mass of the NEP vehicle (not including the spacecraft mass) minus the tank mass. Po is the total power of the vehicle. Tank fraction is also included since it varies **as** a finction of propulsion systems chosen.

TABLE 5. NEP vehicle specific mass.

Power level	0.1 MWe	1 MWe	10 MWe	20 MWe	50 MWe	100 MWe
$M_{NEP(veh\text{-}tk)}$ /Po (kg/kW)	31.7	7.5	3.8	2.7	1.8	1.0
Tank **fraction**	0.5%	OS-2.8%	2.8%	2.8%	2.8%	16%

Ion engines are the most attractive propulsion system (at **25** kW to **500** kW per engine) for robotic missions with system power levels up to a few MWe, while **MPD** thrusters (at **500** kW to ***5*** MW per engine) are very well-suited for **5-100** MWe piloted missions. A large fraction of the mass for the MPD thruster system resides in the power processing and power routing cable mass. Lowering this mass (perhaps with superconducting cables) would significantly improve the net delivered mass.

Recommendations for future work would be to compare these results with other power and propulsion concepts. To make a fair comparison, an effort should be made to assess the implications of those concepts **on** the design of the whole "transportation" vehicle and spacecraft. **To** refine current results, **an** estimate of a sample return lander for sample return missions should be made **and** incorporated in the trades. Also since the *dry* masses of the NEP vehicle are reasonably assessed, they should be incorporated in the trajectory optimizations. Conclusion could then be drawn on the actual optimum specific impulse and delivered masses.

ACKNOWLEDGMENTS

The authors wish to thank John Brophy, Carl Sauer, Joe Lewis, Bob Miyake, and Inam Haque from the Jet Propulsion Laboratory for their tremendous help respectively on thrusters, trajectories, **tanks,** thermal control and high power electronics and cabling. They would also like acknowledge valuable contributions from Mike Houts, Marshall Spaceflight Center, Luis Pinero, Glenn Research Center, Samim Anghaie, INSPI, University of Florida, and Andrea Kodys, Princeton University. Their inputs helped make the detailed analysis possible.

ADDITIONAL TABLES AND FIGURES

100 kW_e Vehicle Mass Breakdown

TABLE 1. 100 kW_e NEP system mass breakdown, and efficiency for 2000 kg of lKrypton

Subsystem components	QTY	Mass /unit (kg)	Total Mass (kg)	Effi-ciency	Comments
Propulsion System			**151.2**		
60-cm Ion Thrusters	5	25.0	125.0	0.77	500 kg throughput/engine
DCIU	2	25	5.0		Scaled-up DS-1 design
Propellant Management System					
Tanks	0.5%	2100	10.5		0.5% of total propellant mass
Feed system fixed	1	1.4	1.4		DS-1 advanced design
Feed system per engine	5	1.5	7.5		DS-1 advanced design
Tubing and fitting	1	1.8	1.8		20% of feed system mass
Power System			**884.9**		
Reactor, heatpipes and controls	1	414.0	414.0		
Radiation shield	1	306.0	306.0		
Power conversion				0.32	
Turbine and compressor	1	18.4	12.0		
Alternator	1	15.0	15.0		
Rotary recuperator	1	10.0	5.0		
Ducting	1	45.0	45.0		
Power processing and management					
PPU	5	8.7	43.5	0.94	Direct-drive design
Power conditioning unit	2	22.2	44.4	0.95	1- kW_e out (1 spare unit)
Thermal control			**622.0**		
Primary radiator (360 K)	1	500.5	500.5		2.3 kg/ kW_{th} Dynatherm
Secondary radiator (300 K)	1	25.0	25.0		3 kg/ kW_{th}, includes PPU radiators
Tanks Sun shade + V-Grove isolation	1	26.5	26.5		
Misc. blankets, heaters, thermostats...	1	70.0	70.0		KBO Team X study
Attitude control			**65.0**		Cold gas + gimbaled momentum wheel
Structures/mechanisms/cabling			**7263**		
IPS structure	26%	146.2	38.0		Not including DCIUs nor tanks
Power structure	16%	856.9	137.1		Includes DCIUs, not reactor/shield
Propellant structure	4%	2100	84.0		
IPS thruster gimbals, actuators, elect.	5	13.5	67.5		Scaled from advanced DS-1 gimbals
PPU micrometeroid shielding	5	1.7	8.5		
Boom	1	206.4	206.4		Scaled version of the AECAble FASTmast
NEP module/S/C interface structure	1	30.0	30.0		KBO Team X study
PPU to thruster cabling	80	0.7	56.0		8 m average/engine + cross-trapped PPUs
PCU to S/C cabling	25	0.2	5.0		25 m
Other cabling	10%	938.1	93.8		Not including reactor nor shield
Subtotal			**2449**		
Contingency (30%)			735		
TOTAL DRY MASS			**3184**		
Deterministic Propellant			2000		
Propellant Residuals, ACS, contingency	5%		100		
TOTAL WET MASS			**5284**		

1 MWe Vehicle Mass Breakdown, Ion Engine and MPDT Propulsion System

TABLE 3. 1MWe Ion engines system mass breakdown, and efficiency for 3800 kg of Krypton

Subsystem components	QTY	Mass /unit (kg)	Total Mass (kg)	Effi-ciency	Comments
Propulsion System			**546.9**		
60-cm Ion Thrusters	18	25.0	450.0	0.75	1900 kg Kr throughput/engine
DCIU	2	2.5	5.0		
Propellant Management System					
Tanks	0.5%	3990	19.9		0.5% of total propellant mass
Feed system fixed	2	3.0	6.0		Twice DS-1 advanced design
Feed system per engine	18	3.0	54.0		Twice DS-1 advanced design
Tubing and fitting	1	12.0	12.0		20% of feed system mass
Power System			**2397.4**		
Reactor, plumbing and controls	1	1215.0	1215.0		SP-100 reactor, 2.4 MWth, 1MWe
Radiation shield	1	700.0	700.0		
Power conversion				0.42	
Turbine and compressor	1	80.0	80.0		
Alternator	1	70.0	70.0		
Rotary recuperator	1	32.0	32.0		
Ducting	1	70.0	70.0		
Power processing and management					
PPU	3	62.0	186.0	0.98	Direct-drive design
Power conditioning unit	2	22.2	44.4	0.95	1-kW_e out (1 spare unit)
Thermal control			**1025.2**		
Primary radiator (450 K)	1	769.6	769.6		0.55 kg/ kW_{th}
Secondary radiator (300 K)	1	115.0	115.0		2.3 kg/ kW_{th}, includes PPU radiators
Tanks Sun shade + V-Grove isolation	1	40.6	40.6		
Misc. blankets, heaters, thermostats...	1	100.0	100.0		KBO Team X study scaled
Attitude control			**130.0**		Cold gas + gimbaled momentum wheel
Structures/mechanisms/cabling			**1697.3**		
IPS structure	26%	541.9	140.9		Not including DCIUs nor tanks
IPS thruster deployment structure	1	341.3	341.4		Half of thrusters+feed system+ IPS struct.
Power structure	16%	1642.6	262.8		Includes DCIUs, not reactor/shield
Propellant structure	4%	3990	159.6		
IPS thruster gimbals, actuators, elect.	18	13.5	243.0		Advanced DS-1 gimbals
Boom	1	301.0	301.0		Scaled version of the AECAble FASTmast
NEP module/S/C interface structure	1	30.0	30.0		KBO Team X study
Alternator to PPU cabling	8	0.57	4.6		8 m
PPU to thruster cabling	18	2	36.0		4 m average/engine + cross-trapped PPUs
PCU to S/C cabling	30	0.2	6.0		25 m
Other cabling	10%	1734.6	173.5		Not including reactor nor shield
Subtotal			**5797**		
Contingency (30%)			1739		
TOTAL DRY MASS			**7536**		
Deterministic Propellant			3800		
Propellant Residuals, ACS, contingency	5%		190		
TOTAL WET MASS			**11526**		

TABLE 4. 1 MWe LFA system mass breakdown, and efficiency for 3800 kg of Litbium

Subsystem components	QTY	Mass /unit	Total Mass	Efficiency	Comments All masses in kg
Propulsion System			**488.2**		
Li LFA (anode, cathode, vaporizer)	3	116.2	348.7	**0.6**	500-kW each, 4500 kg throughput/engine
DCIU	2	2.5	**5.0**		
Propellant Management System					
Tanks	2.8%	111.7	111.7		2.8% of total propellant mass
Feed system fixed	1	4.0	4.0		
Feed system per engine	3	**5.0**	15.0		
Tubing and fitting	1	3.8	3.8		20% **of** feed system mass
Power System			**4579.4**		
Reactor, plumbing and controls	1	1215.0	1215.0		
Radiation shield	1	700.0	700.0		
Power conversion				0.42	
Turbine and compressor	1	80.0	80.0		
Alternator	1	70.0	70.0		
Rotary recuperator	1	32.0	32.0		
Ducting	1	70.0	70.0		
Power processing and management			***1091.4***		
Transformer & radiator	1	500.0	500.0		
CR/F Input switches	12	20.5	246.0		
Controlled rectifiers (CR)	3	34.0	102.0	0.97	
Filters (F)	3	20.0	60.0	0.99	
CR/F Output switches	2	28.5	57.0		
Thruster switches	2	41.0	82.0		
Power conditioning unit	2	22.2	44.4	0.95	1-kW_e out (1 spare unit)
Cabling			***1204.0***		
Turboalternator to transformer	**50**	0.57	28.5		**50** m at 0.57 kg/m (high **V**, low A)
Transformer to PMAD	1	0.57	0.6		1 m at 0.57 kg/m (high **V**, low A)
Input to switch to CR	**9x2**	20.0	360.0		2 m at 20 kg/m (low **V**, high A)
Input to spare **CR** switch	6	20.0	120.0		1 m at 20 kg/m (low **V**, high A)
CR internal	9	20.0	180.0		**1** m at 20 kg/m (low **V**, high A)
CR to filter to switch to output	6x0.5	30.5	91.5		**0.5** m at 30.5 kg/m (low **V**, high A)
Output parallel connections	2	30.5	61.0	0.97	**0.5** m at 30.5 kg/m (low **V**, high **A)**
Current leads PPU to thruster	4	30.4	121.6		1 m **at** 30.5 kg/m (low **V**, high A)
Cross-members, insulation...	0.25	963.2	240.8		
Thermal control			**1004.5**		
Primary radiator (450 **K**)	1	769.6	769.6		**0.55** kg/ kW_{th}
Secondary radiator (300 **K**)	1	134.9	134.9		2.3 kg/ kW_{th}
Misc. blankets, heaters, thermostats**...**	**1**	100.0	100.0		KBO Team X study scaled
Attitude control			**130.0**		Cold **gas** + gimbaled momentum wheel
Structures/mechanisms/cabling			**1498.4**		
IPS structure	26%	488.2	126.9		Not including DCIUs nor tanks
Power structure	16%	2518.9	403.0		Includes DCIUs, not reactor/shield
Propellant structure	4%	3990.0	159.6		
IPS thruster gimbals, actuators, elect.	3	65.1	195.4		Scaled from advanced DS-1 gimbals
Boom	1	301.0	**301.0**		Scaled version **of** the AECAble FASTmast
NEP module/S/C interface structure	**1**	30.0	30.0		KBO Team **X** study
Plume shield	1	***95.0***	95.0		
Other cabling	10%	1874.7	187.5		Not including reactor nor shield
Subtotal			**7701**		
Contingency **(30%)**			2310		
TOTAL DRY MASS			**10011**		
Deterministic Propellant			3800		
Propellant Residuals, ACS, contingency	***5%***		190		
TOTAL WET MASS			**14001**		

100 kW$_e$ Pluto Rendezvous Results

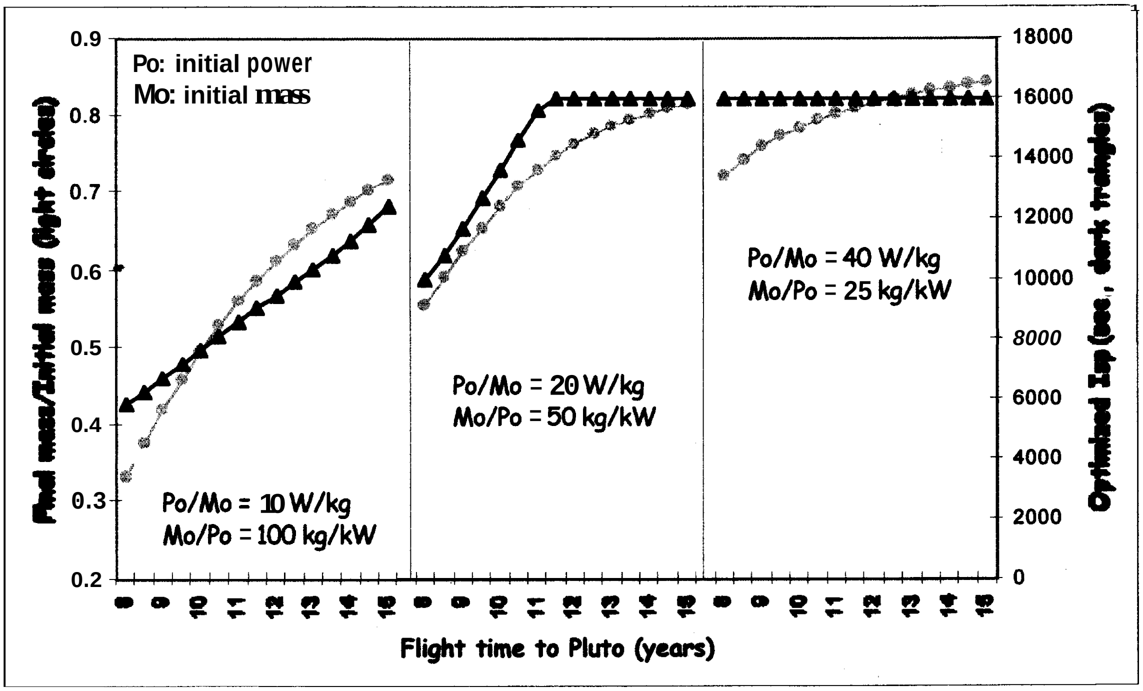


FIGURE 6. 100 kW$_e$ Pluto rendezvous mission trajectory results.'

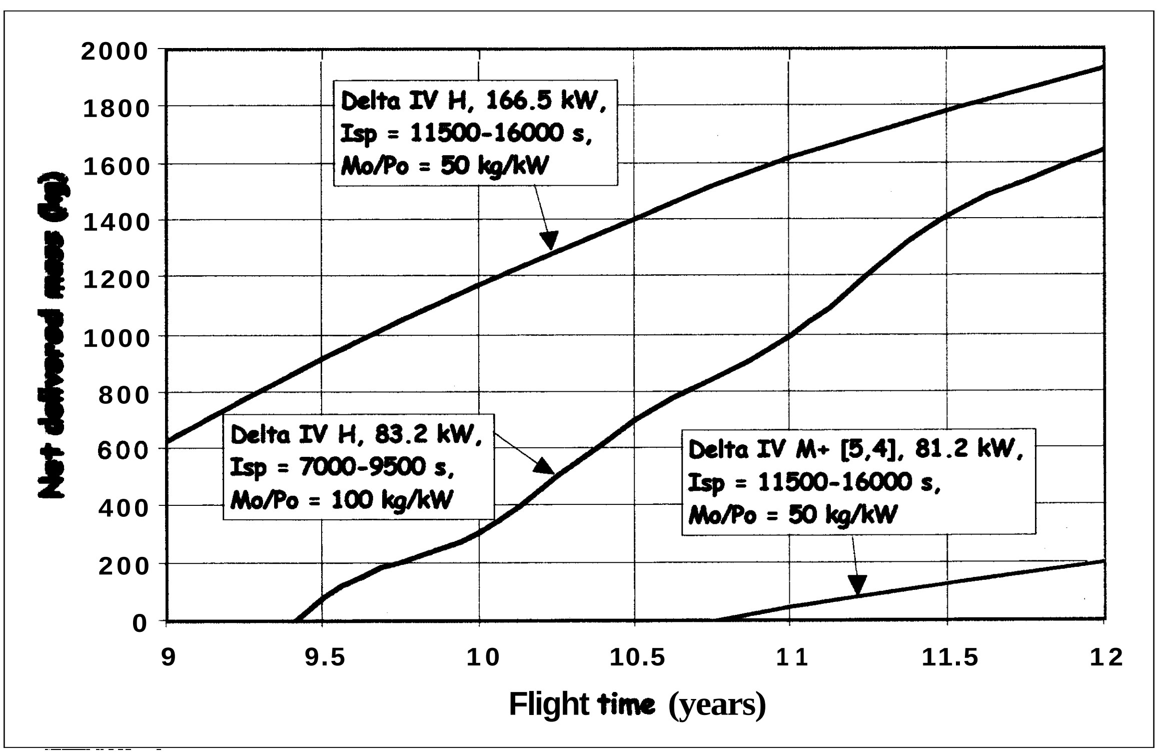


FIGURE 7. Net delivered mass for the Pluto rendezvous mission as a function of flight time. 100 kW$_e$ vehicle.

100 kW$_e$ TitadSaturn Sample Return Results

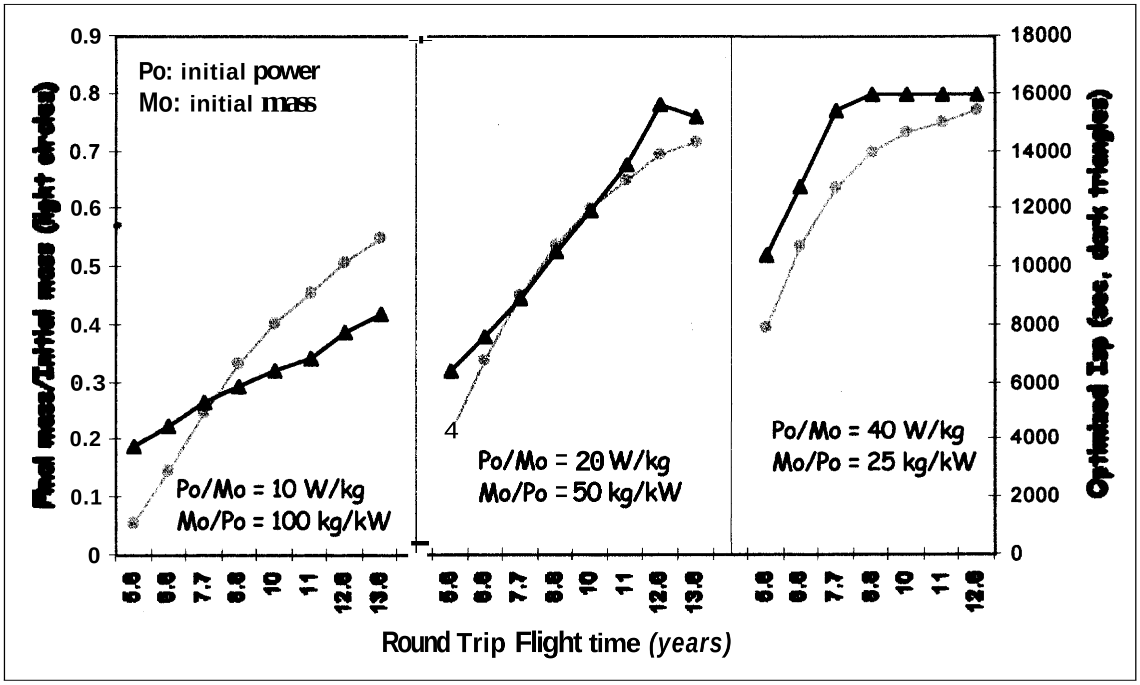


FIGURE 8. 100 kW$_e$ Titan/Saturn sample return trajectory results.

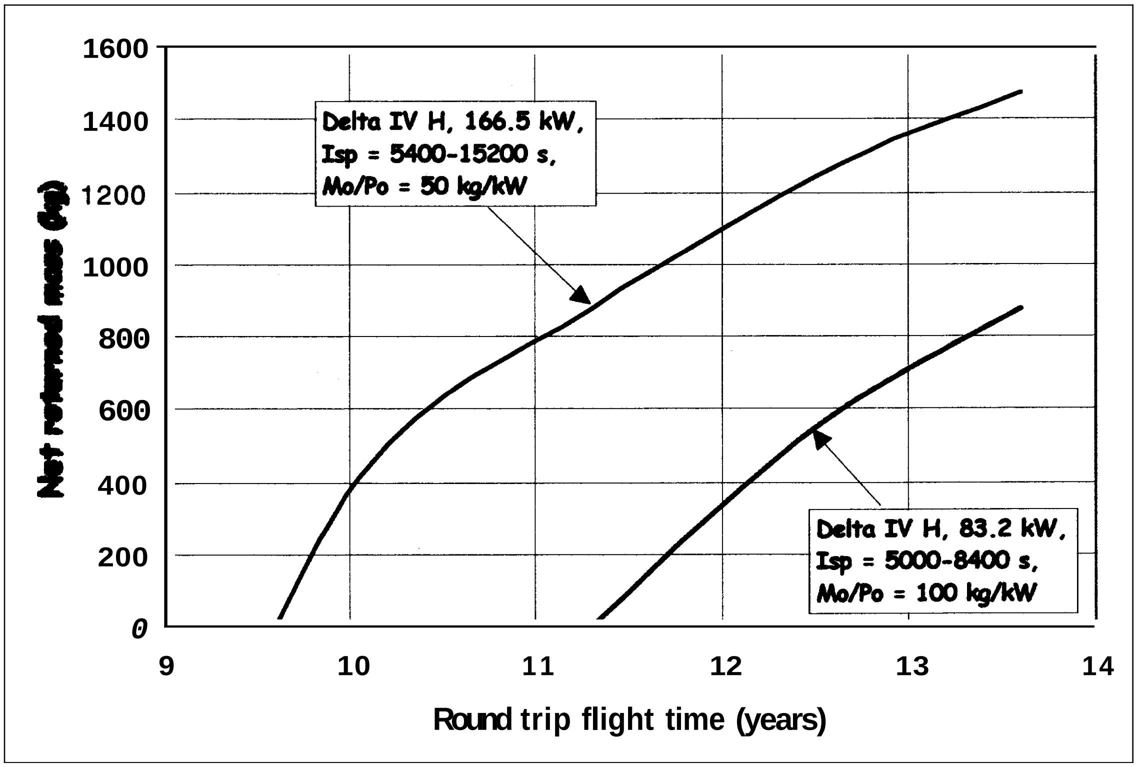


FIGURE 9. Net returned mass for the TITAN/SATURN Sample Return as a function of flight time. 100 kW$_e$ vehicle.

100 kW$_e$ Europa Orbiter Results

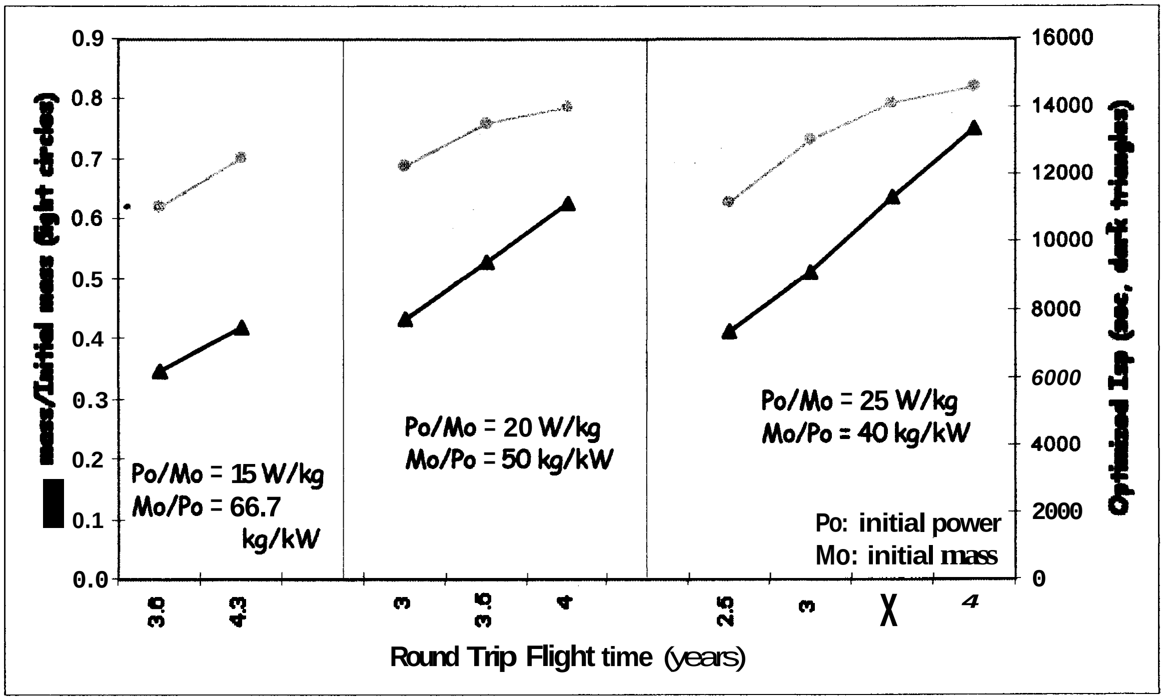


FIGURE 10. 100 kW$_e$ **Europa** orbiter trajectory results.

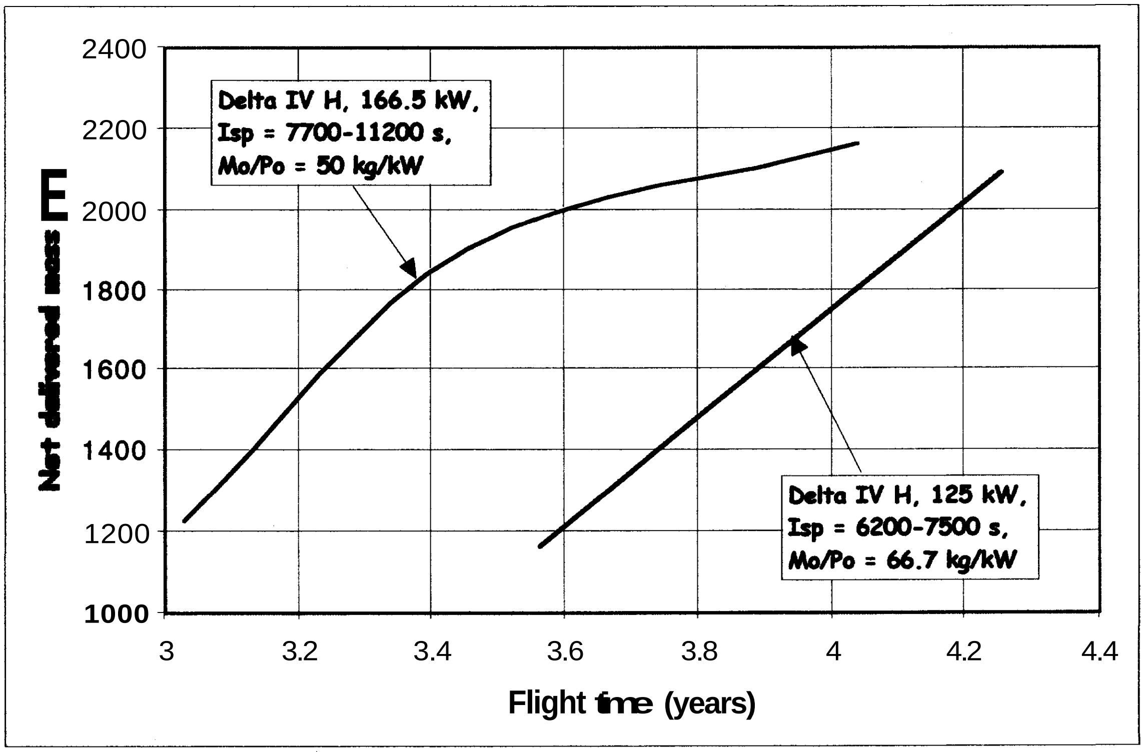


FIGURE 11. Net delivered mass **for** the **EUROPA** Orbiter **as a** function **of** flight time. 100 kW$_e$ vehicle

1 MWe TitadSaturn Sample Return Results

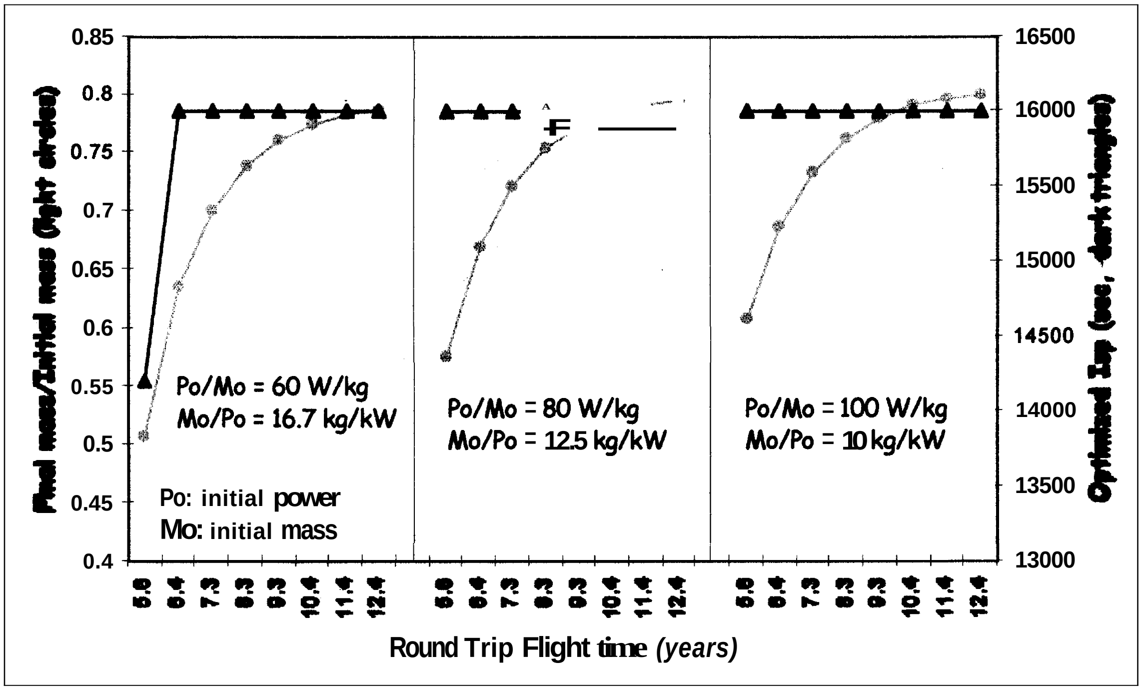


FIGURE 15.1 MWe TITAN/SATURN Sample Return trajectory results.

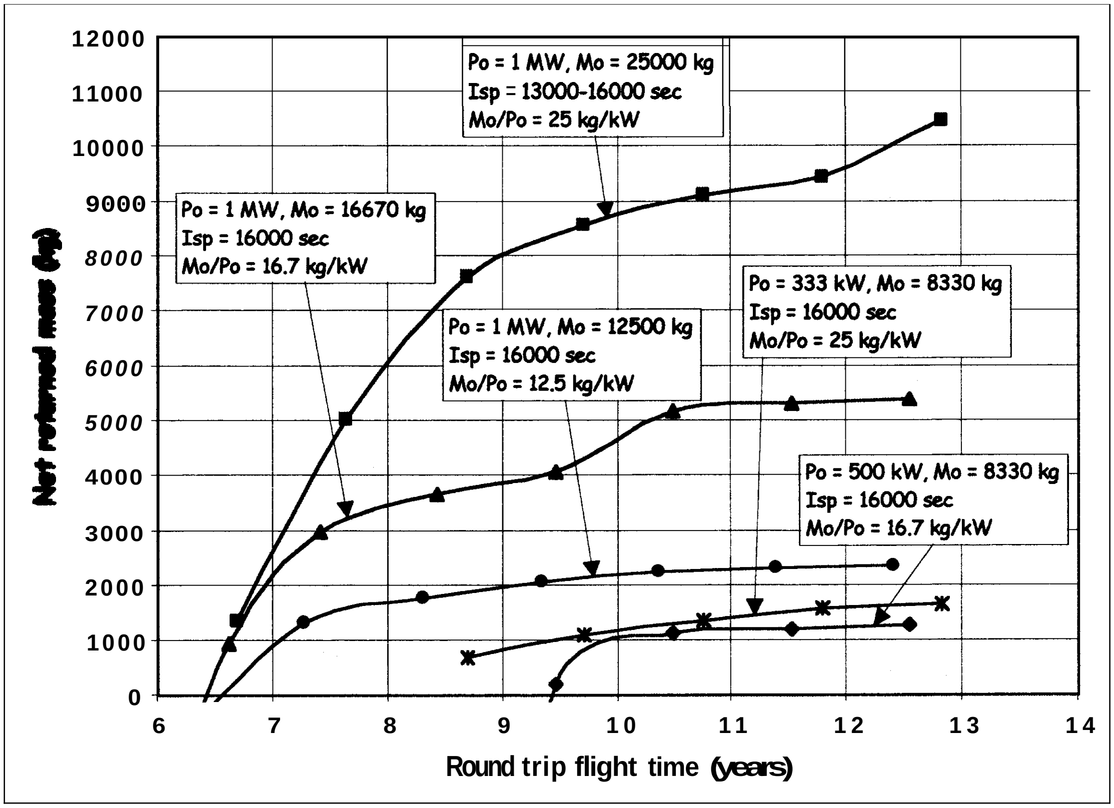


FIGURE 16. Net returned mas for the TITAN/SATURN Sample Return as a function of flight time. 1 MWe vehicle.

1 MWe Europa Sample Return Results

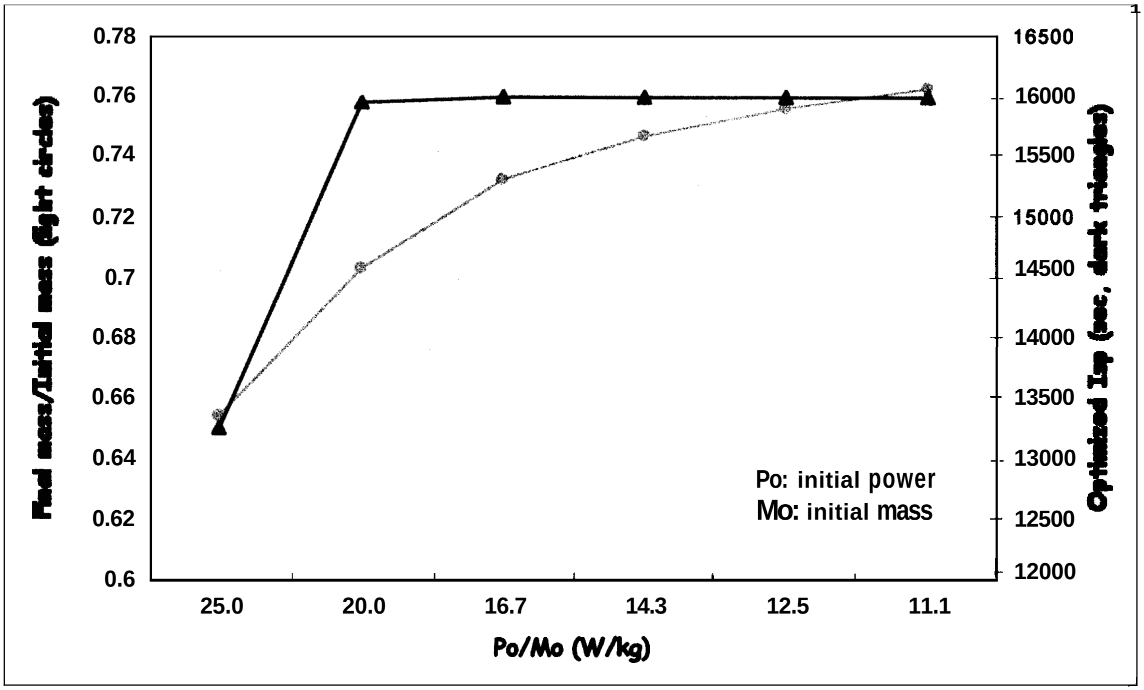


FIGURE 17. 1 MWe Europa sample return trajectory results.

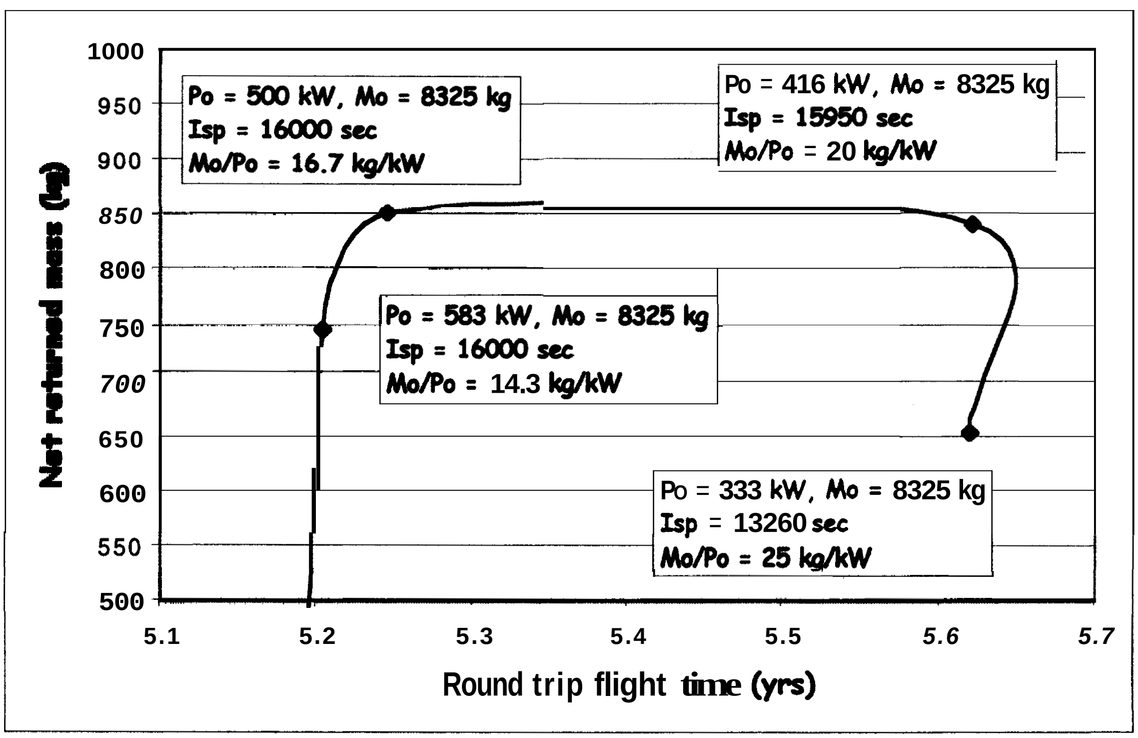


FIGURE 18. Net returned mass for the Titan/Saturn sample return as a function of flight time. 1 MWe vehicle.

Multi-MW Mars Piloted Mission Results

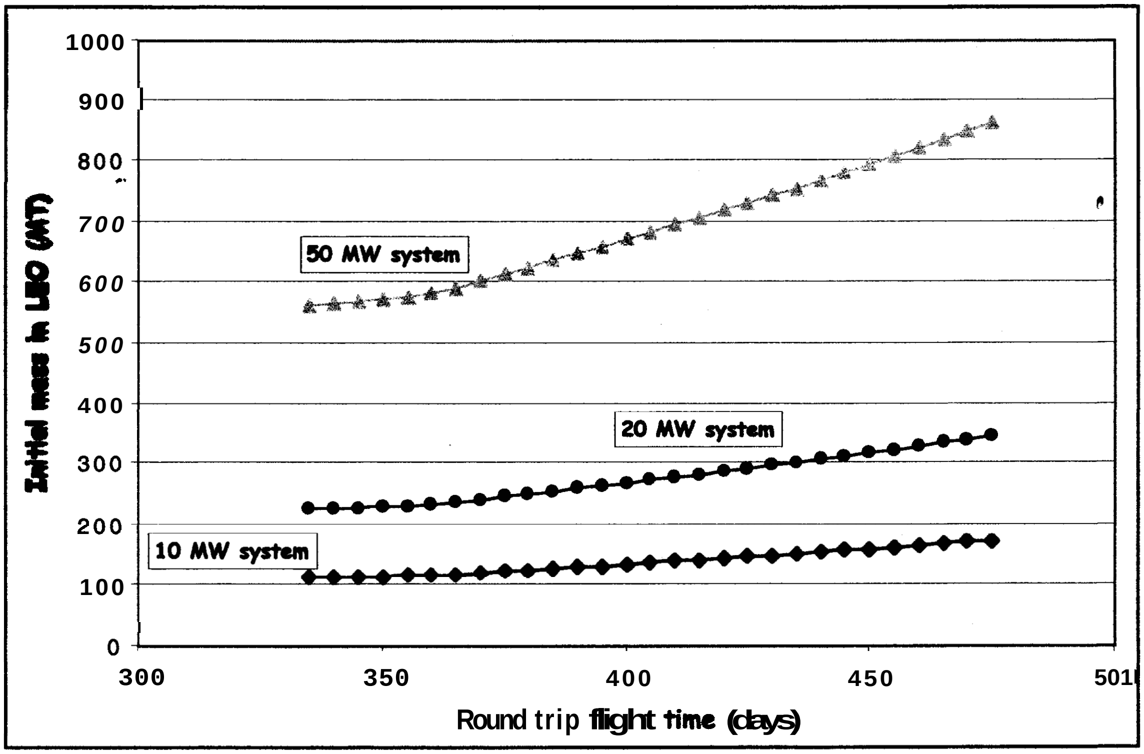


FIGURE 19. **1** MWe Mars piloted mission results – **IMLEO as a** function **of** round trip time and power level.

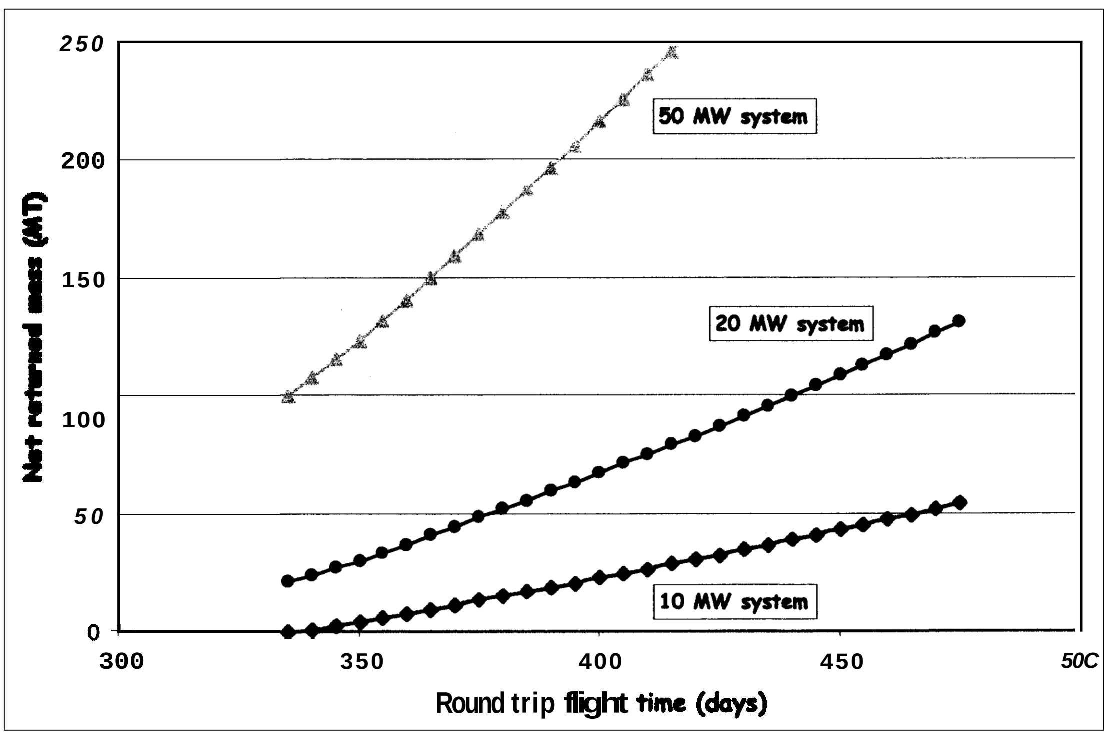


FIGURE 20. **1** MWe Mars piloted mission results – net returned **mass as a** function **of** round **trip** time and power level.

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Samanta Roy, **R.**

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STAIF 2001

18th Symposium on Space Nuclear Power and Propulsion

Evolutionary Strategy for the Use of Nuclear Elec ric Propulsion in Planetary Exploration

Feb. 13, 2001

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Introduction

NASA

- Acknowledgments
- Study Objectives:

  (Re-)Evaluate performances of NEP for various missions and power levels

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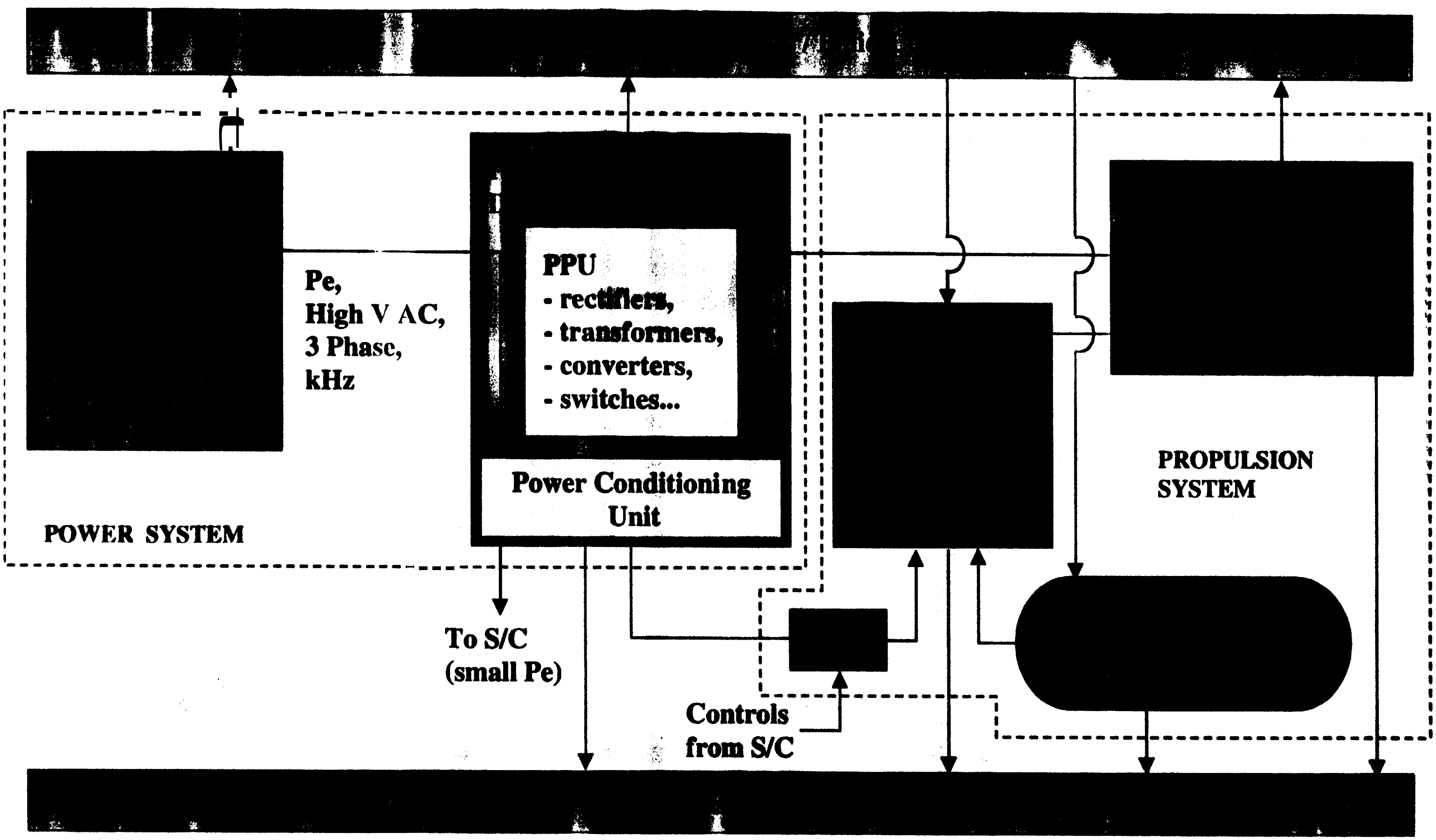
Approach

- Design NEP vehicle (not just power or propulsion systems) with as much detail as possible
- Consider four power levels that constitute an evolutionary path for the technology:

Power level	100-kWe	0.3-1 MWe	10-50 MWe	100 MWe
Mission type	Robotic Orbiter	Robotic SR	Human Inner Pl.	Human Outer Planet
Power System	SAFE-300 Recup. Brayton	SP-100 technology Brayton		Fission. plasma core MHD/closed Rankine
Propulsion	25-kW ion	480-kW ion	1-5 MW MPDTs	> 5MW-MPDTs

NEP System Level Block Diagram



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Power System Assumptions

NASA

Power levels	100-kWe	1-50 MWe	*100 MWe
Technology	SAFE-300	XP/Gas Cooled	Fiss. Gas/Molten Core
Nuclear Fuel	UN/UO_2	UN/UO_2	UO_2 UF_4
Thermal power	320 kWth	2.4-100+ MWth	TBD
Electric power	100-kWe	1-50 MWe	100 MWe
Conversion Efficiency	32%	42%	>40%
Conversion cycle	Recup. Brayton	Recup-Brayton	MHD/Brayto Rankine
Thermal radiator type	CPL-LHP	CPL-LHP-LDR?	LDR-dep.she
Radiator specific mass	2.3 kg/kWth	0.55 kg/kWth	<.1 kg/kWth
	4 kg/m²	2.3 kg/m²	<0.5kg/m²
Two sided radiating area	130 m²	3[illegible]0 m²	very large
Radiator temperature	360 K	450 K (cte temp)	~450K
Power syst. spec. mass	18 kg/kWe	4.3-4.6 kg/kWe	0.5 kg/kWe

(includes power generation, processing (PPUs), radiators)

*Requires Detailed Study Tech Options Only

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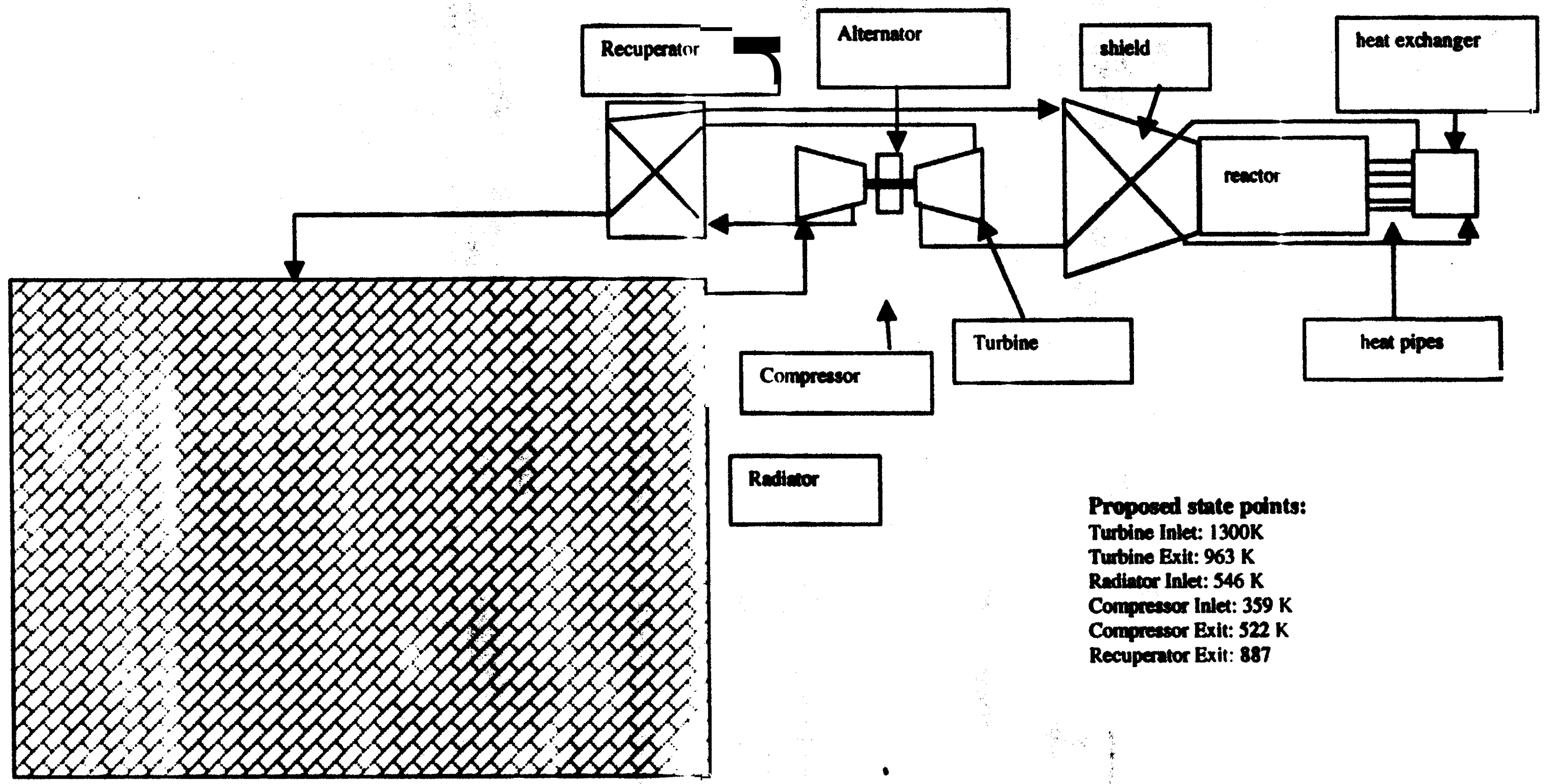
100-kWe Power System Description



Proposed state points:
Turbine Inlet: 1300K
Turbine Exit: 963 K
Radiator Inlet: 546 K
Compressor Inlet: 359 K
Compressor Exit: 522 K
Recuperator Exit: 887

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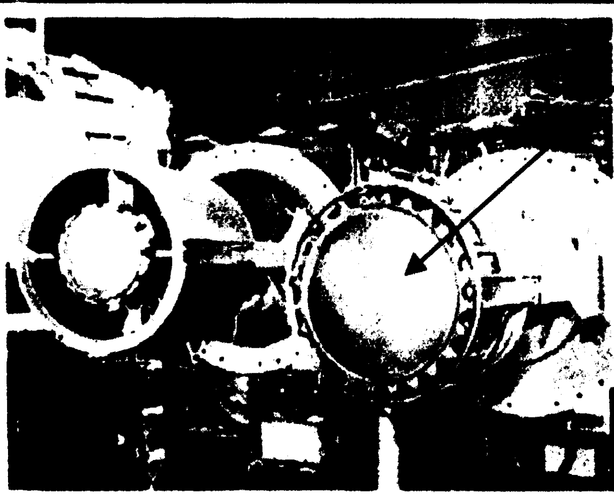
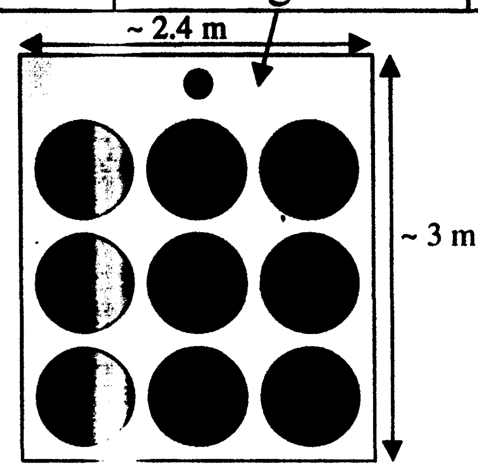
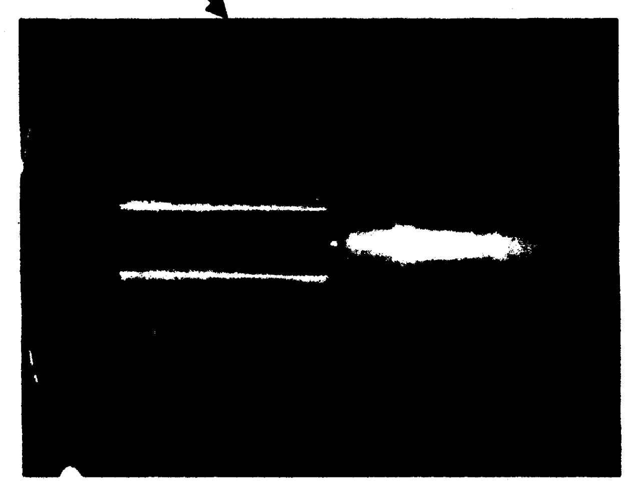
Propulsion System Assumptions

NASA

Mission power level	100-kWe	1 MWe	10-50 MWe	100 MWe
Engine type	25-kW ion	[illegible]kW ion	0.5-MW MPDT	MPDT
Propellant	Kr	[illegible]	Li	D_2
Efficiency	0.77	0.77	0.6	0.65
Specific Impulse	16000 s	16000 s	5000 s	15000 s
Mass	25 kg	225 kg	116	-
Prop. throughput/eng.	500 kg	19[illegible] kg	4500 kg	-
Tankage fraction	0.5%	[illegible] 5%	2.8 %	16 %
Propellant storage	Liq. 120 K	Liq. 120 K	Solid	Liq. Cryo.
PPU	Direct drive	Direct drive	Transf. +P[illegible]	-
Propulsion spec. mass	2 kg/kWe	0.7 kg/kWe	[illegible] 6 kg/kWe	







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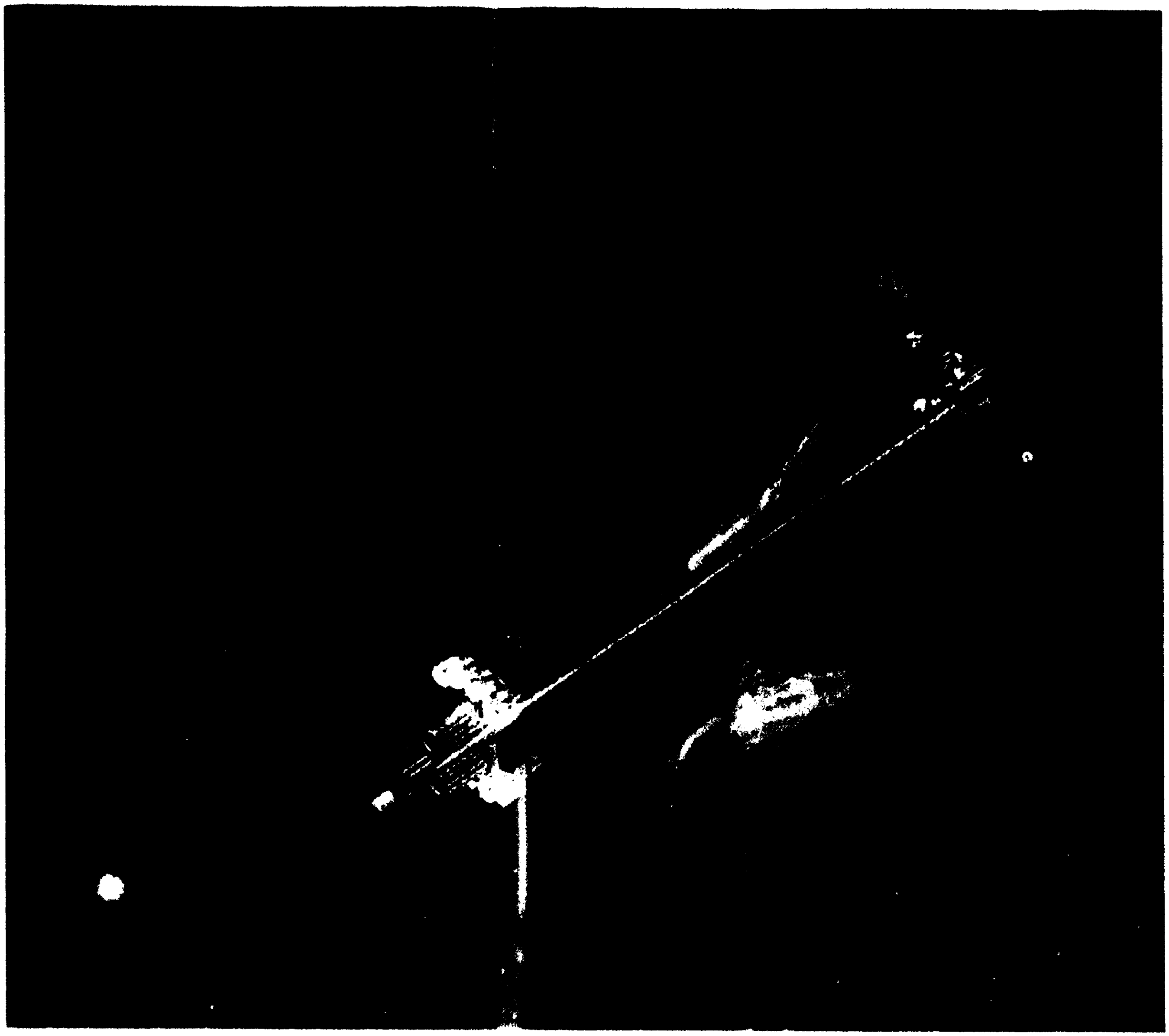
0.1-1 MWe Ion Propulsion Vehicle Configuration



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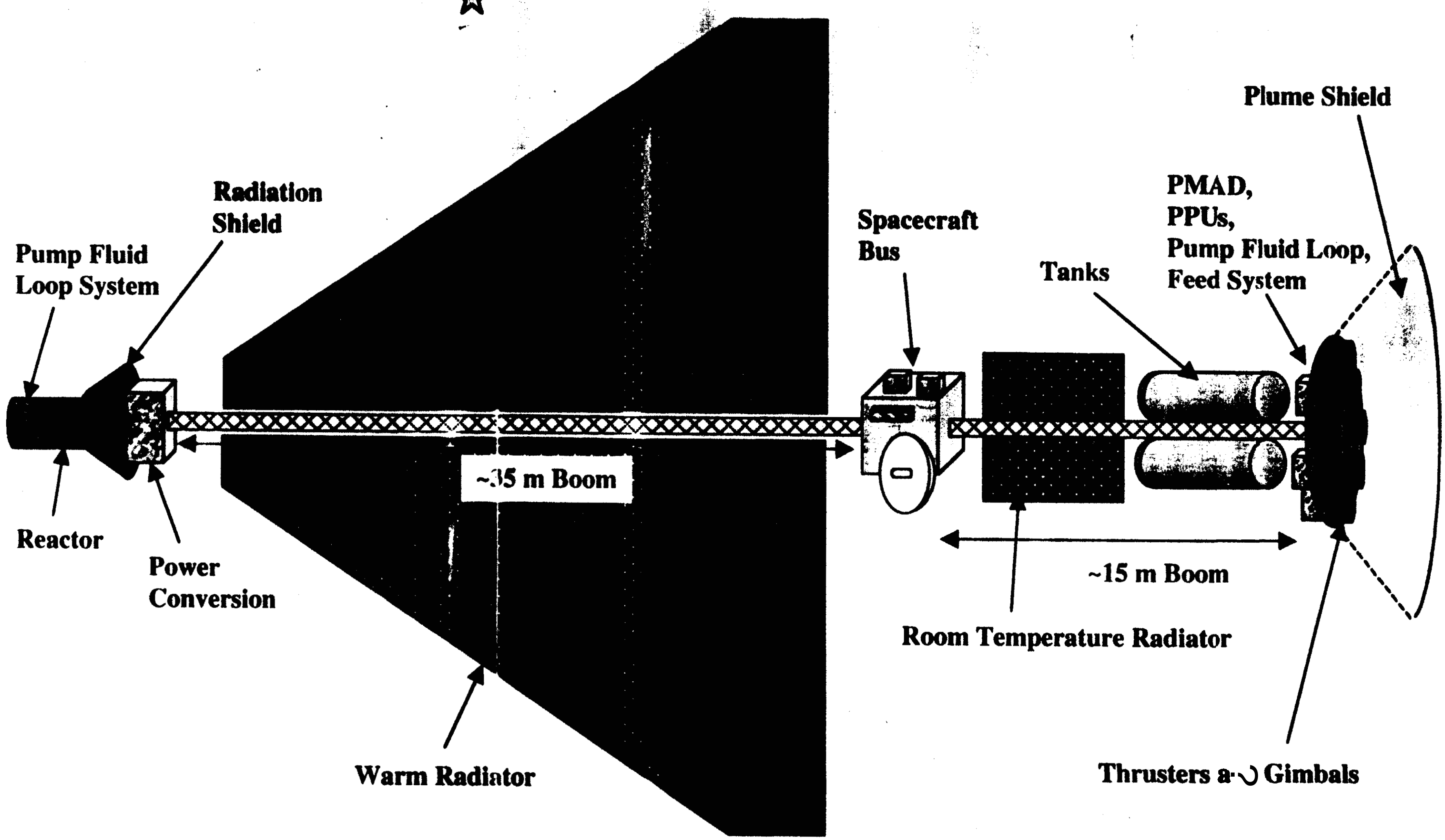
10-50 MWe MPDT Vehicle Configuration



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Other Subsystem Assumptions

- Propulsion redundancy rules:
  - 1 spare ion engine per 4 operating
  - 1 spare MPDT
  - spare PPU and DCIU (single fault tolerance)
- Structures subsystem:
  - Consistent with JPL Team X design guideline:
    - 26% of propulsion system
    - 4% of propellant
    - 16% of all other subsystem
  - Cabling: 10% of most subsystem
- Thermal subsystem
  - 3-4% of total dry in addition to Thermal system inherent to the Power System
- Attitude Control
  - Cold gas system + reaction wheels (probably not optimum)
- Mechanisms/Boom
  - Deployment boom mass assume similar design to the AEC-Able FastMast for ISS with 20-30% mass reduction
  - IPS deployment mechanism for 480-kW ion engine
- Stowed configuration roughly fit within 5-m diameter fairing (Delta IV)
- 30% overall system contingency

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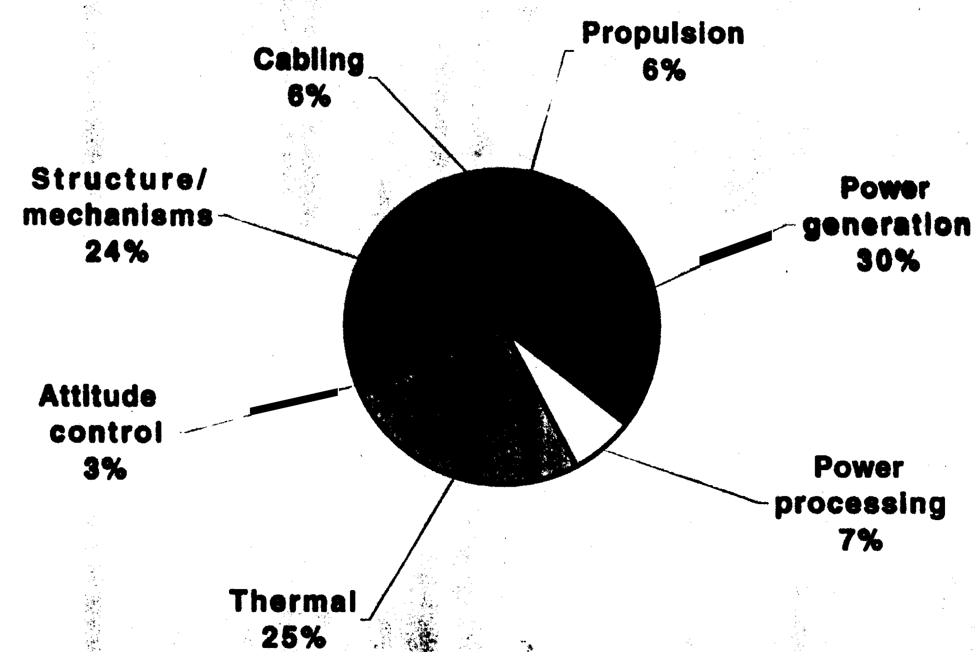
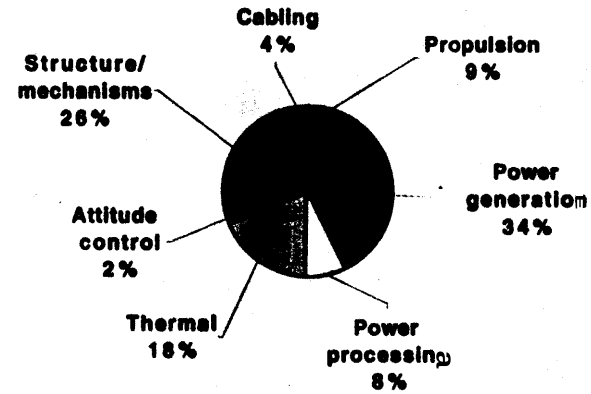
100-kWe NEP Vehicle Mass Breakdown

Propulsion	**151**	**kg**
- Thrusters	125	
- PMS	21	
- DCIU	5	
Po r	**885**	**kg**
- Reactor, HP, controls	414	
- Radiation shield	306	
- Power conversion	77	
- Power processing	88	
Thermal co trol	**622**	**kg**
- Primary radiator	501	
- Other	121	
ACS	**65**	**kg**
Structures/Mech./Cabling	**726**	**kg**
- IPS + prop. Structure	198	
- Power structure	137	
- Boom	206	
- NEP/S/C interface	30	
- Cabling	155	
Subtotal	**2449**	**kg**
Contingency (30%)	**735**	
Total Dry Mass	**3184**	**kg**
Fluids	**2100**	**kg**
Total Wet Mass	**5284**	**kg**

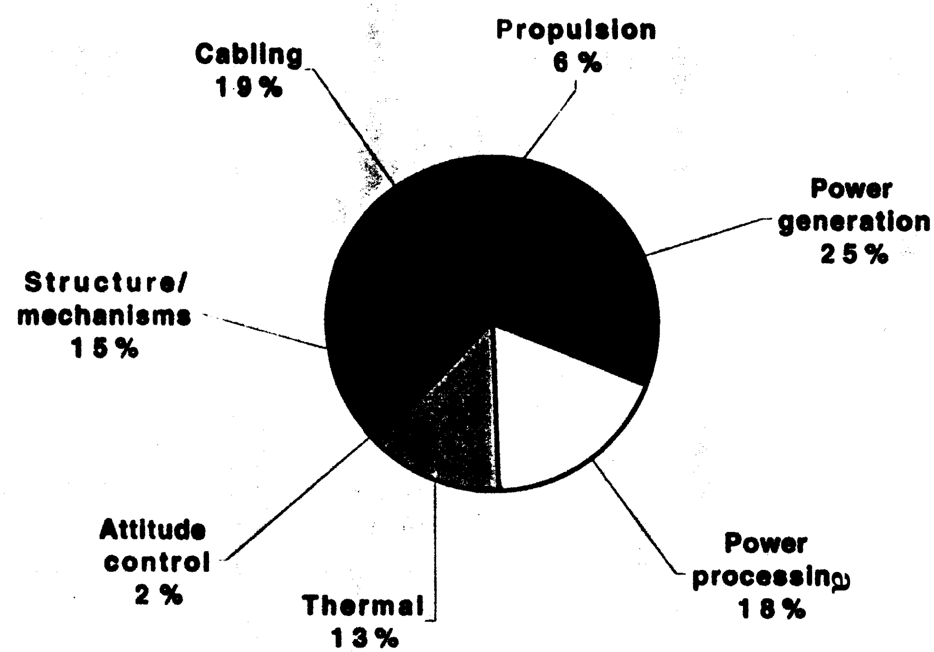


1-MWe NEP Vehicle Mass Breakdown

Propulsion (ion thrusters)	**547**	**kg**
- Thrusters	450	
- PMS	92	
- DCIU	5	
Power	**2397**	**kg**
- Reactor, HP, controls	1215	
- Radiation shield	700	
- Power conversion	252	
- Power processing	230	
Thermal control	**1025**	**kg**
- Primary radiator	770	
- Other	255	
ACS	**130**	**kg**
Structures/Mech./Cabling	**1697**	**kg**
- IPS + prop. Structure	885	
- Power structure	263	
- Boom	301	
- NEP/S/C interface	30	
- Cabling	220	
Subtotal	**5797**	**kg**
Contingency (30%)	**1739**	
Total Dry Mass	**7536**	**kg**
Fluids	**3990**	**kg**
Total Wet Mass	**11526**	**kg**



MPDT propulsion vehicle total dry mass: 10011 kg (including 30% contingency)



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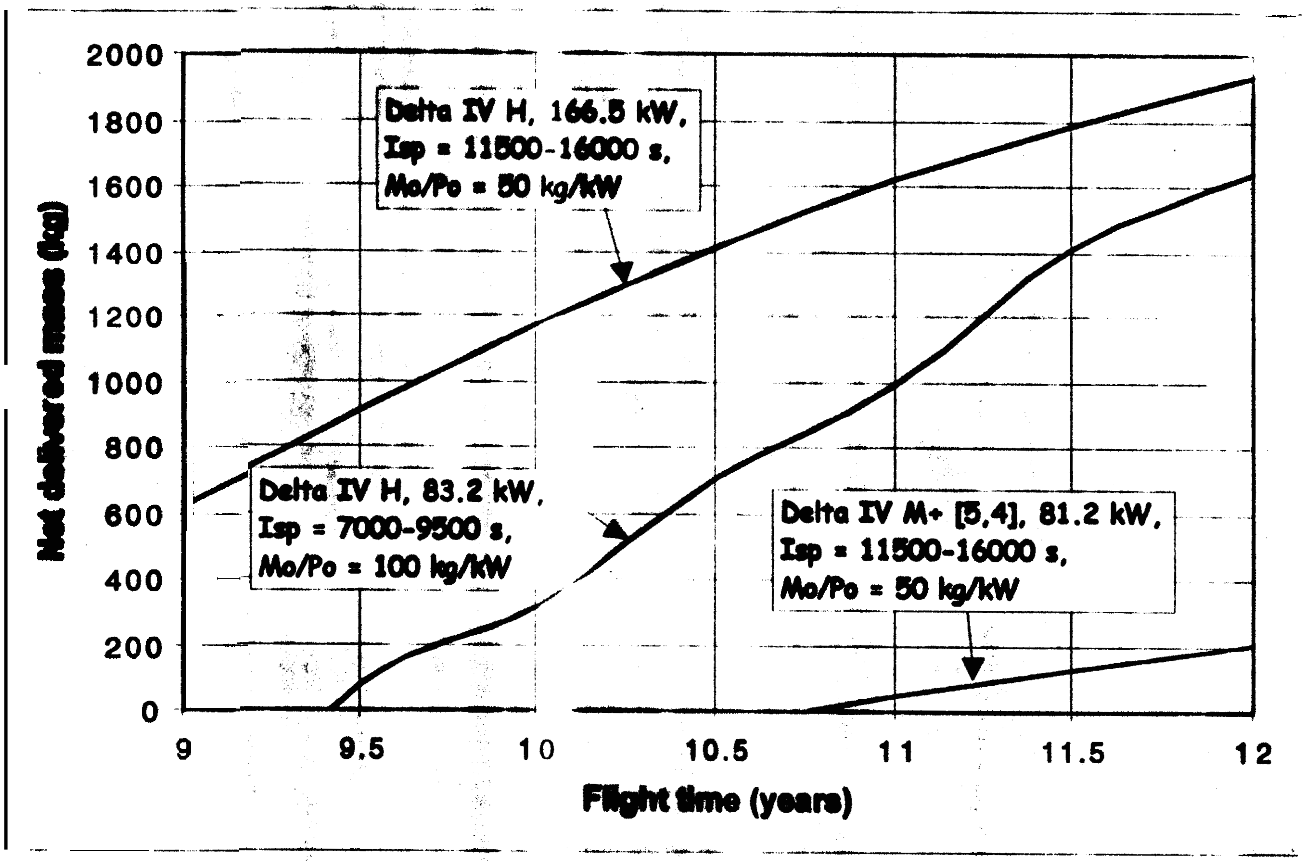
100-kWe: Mission Results

Pluto Orbiter Mission



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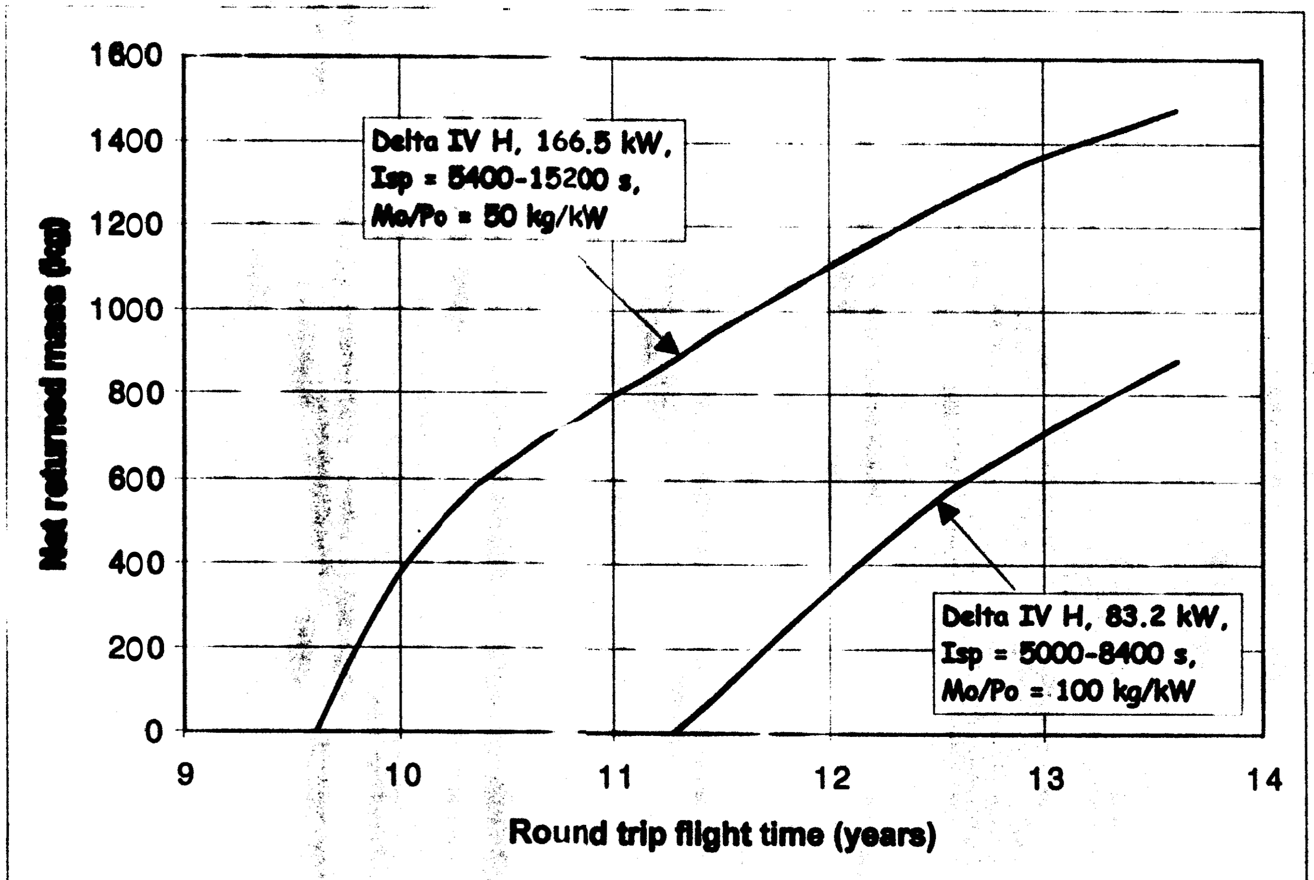
100-kWe: Mission Results

Saturn/Titan Sample Return Mission



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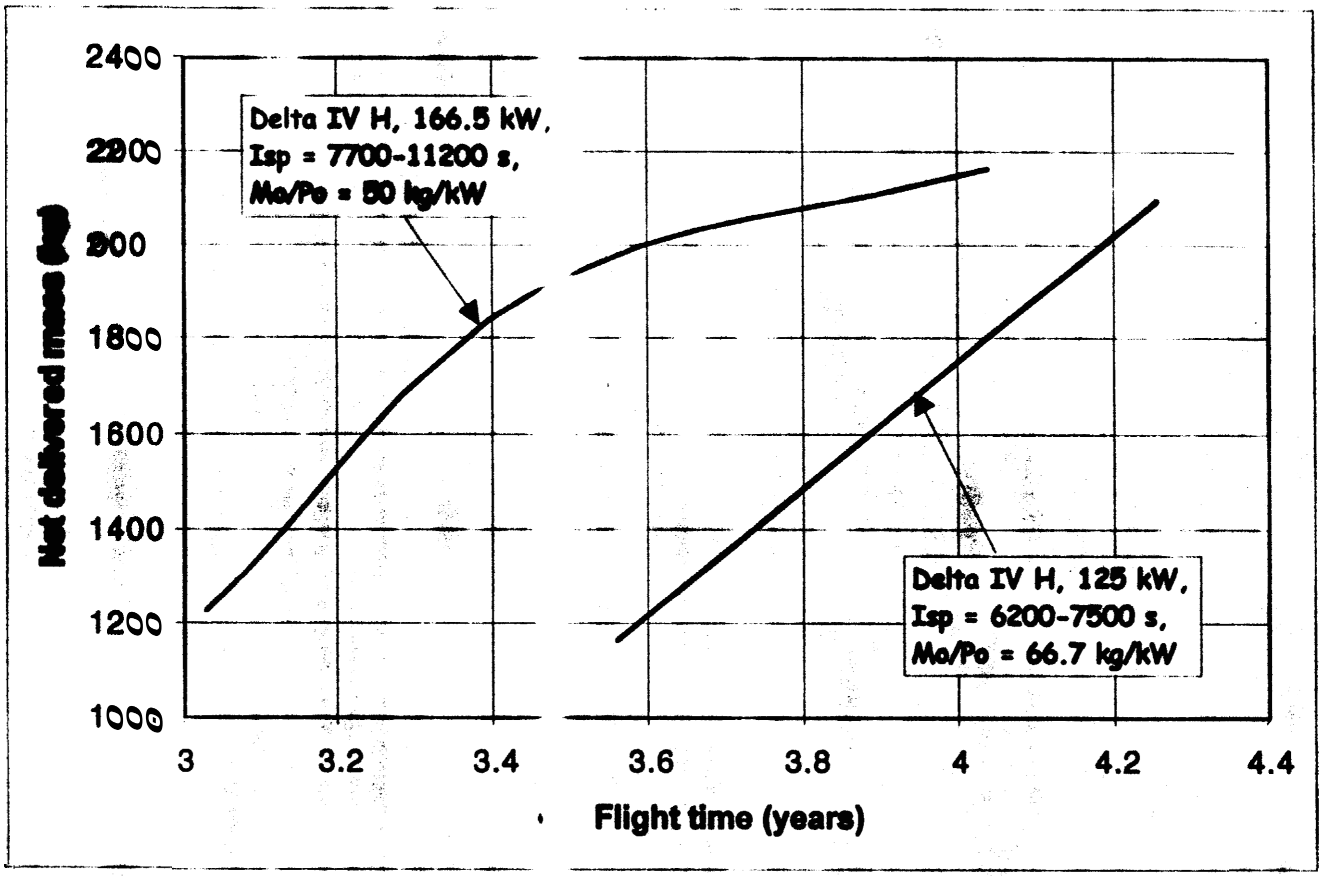
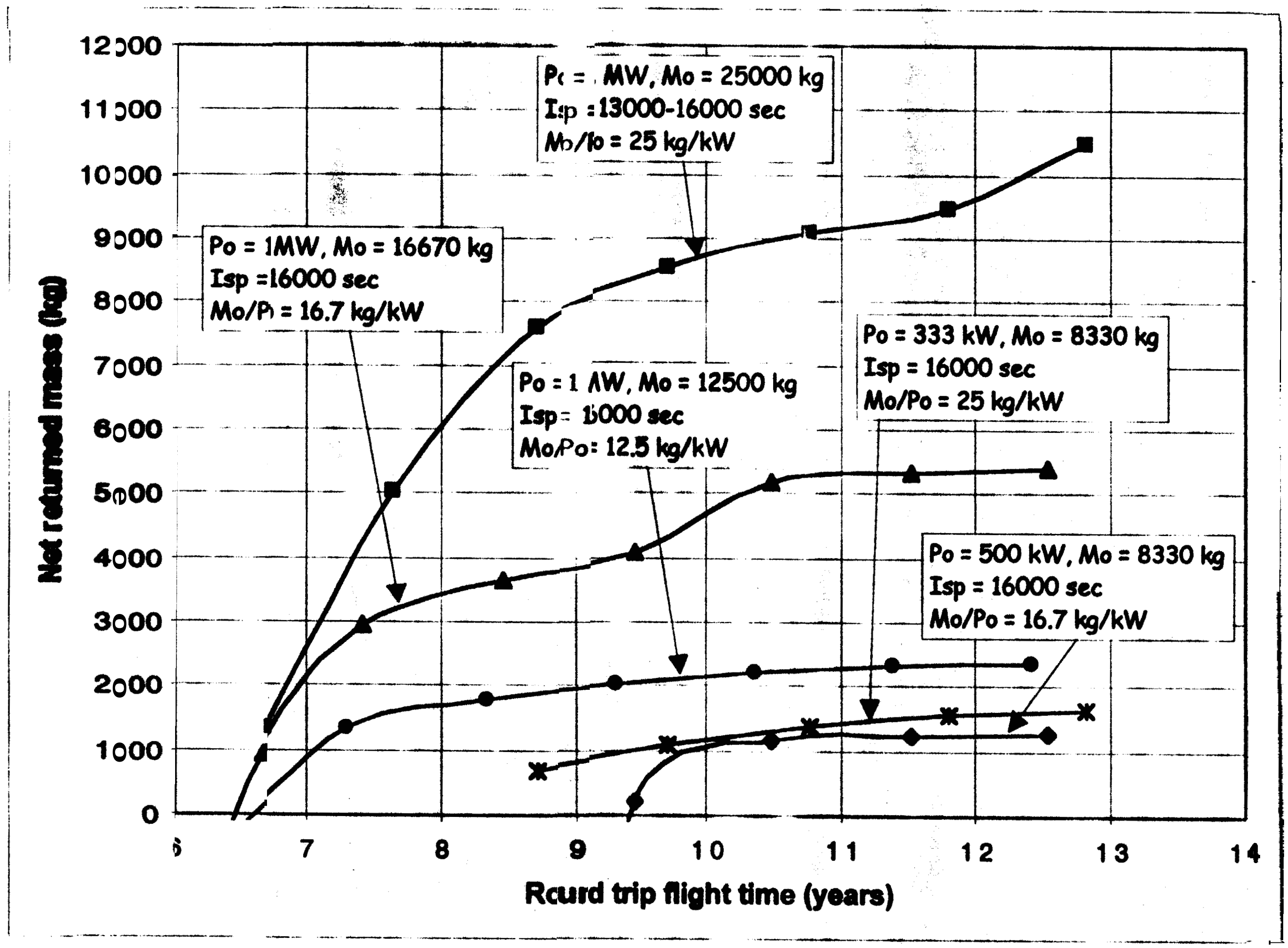
100-kWe: Mission Results

Europa Orbiter/Lander Mission



3-1 MWe: Mission Results

Saturn/Titan Sample Return



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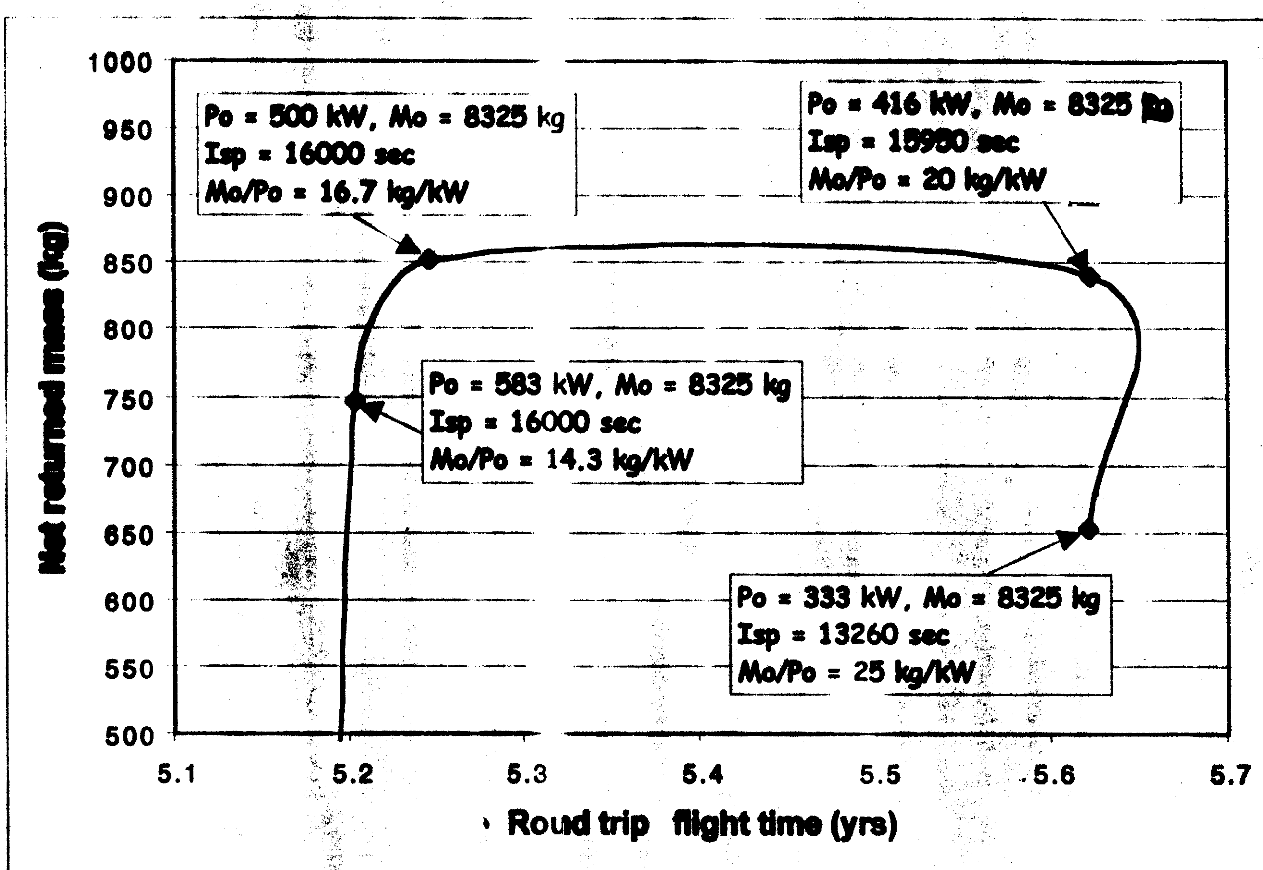
3-1 MWe: Mission Results

Europa Sample Return



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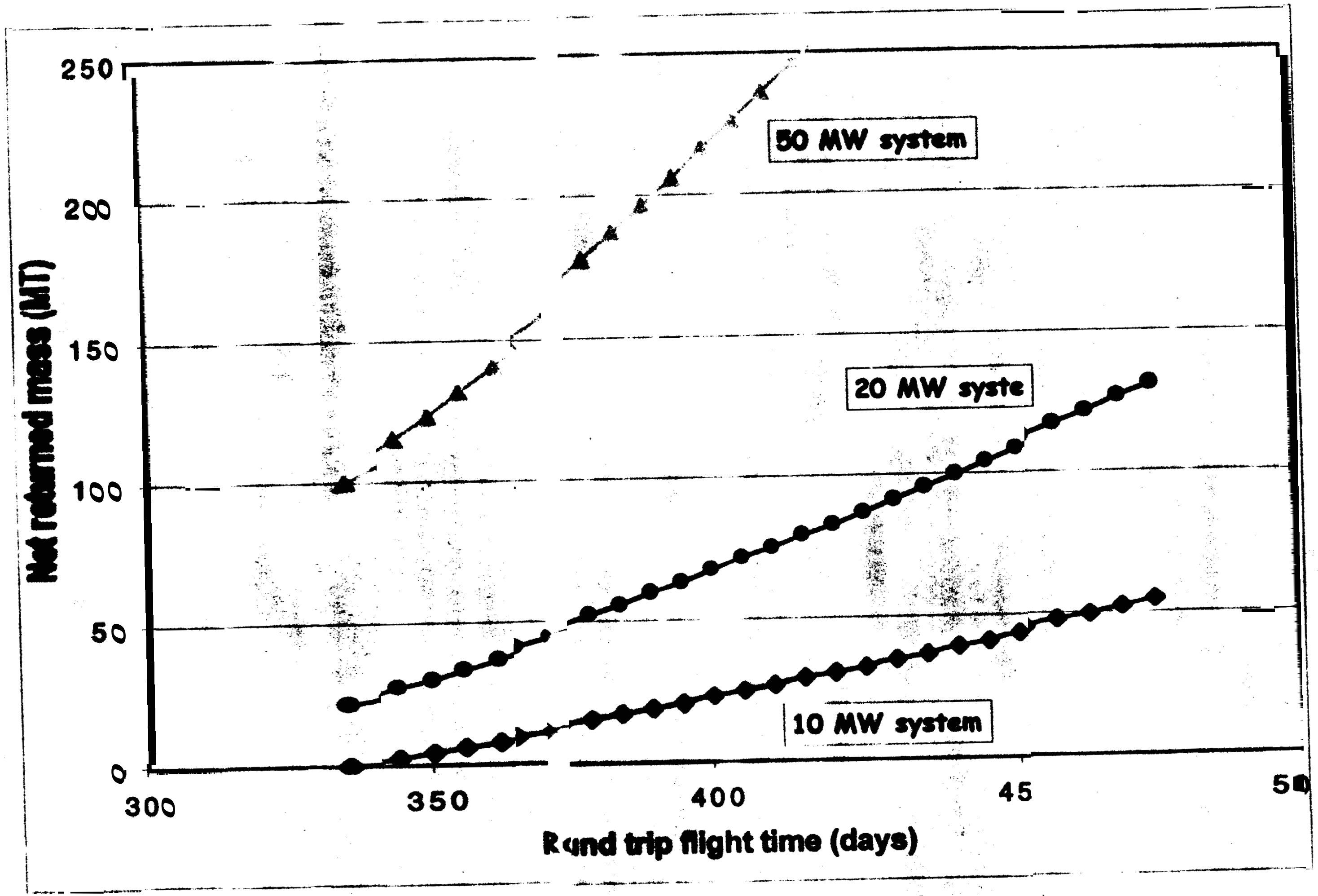
10-50 MWe Mission Results

Mars Piloted Mission



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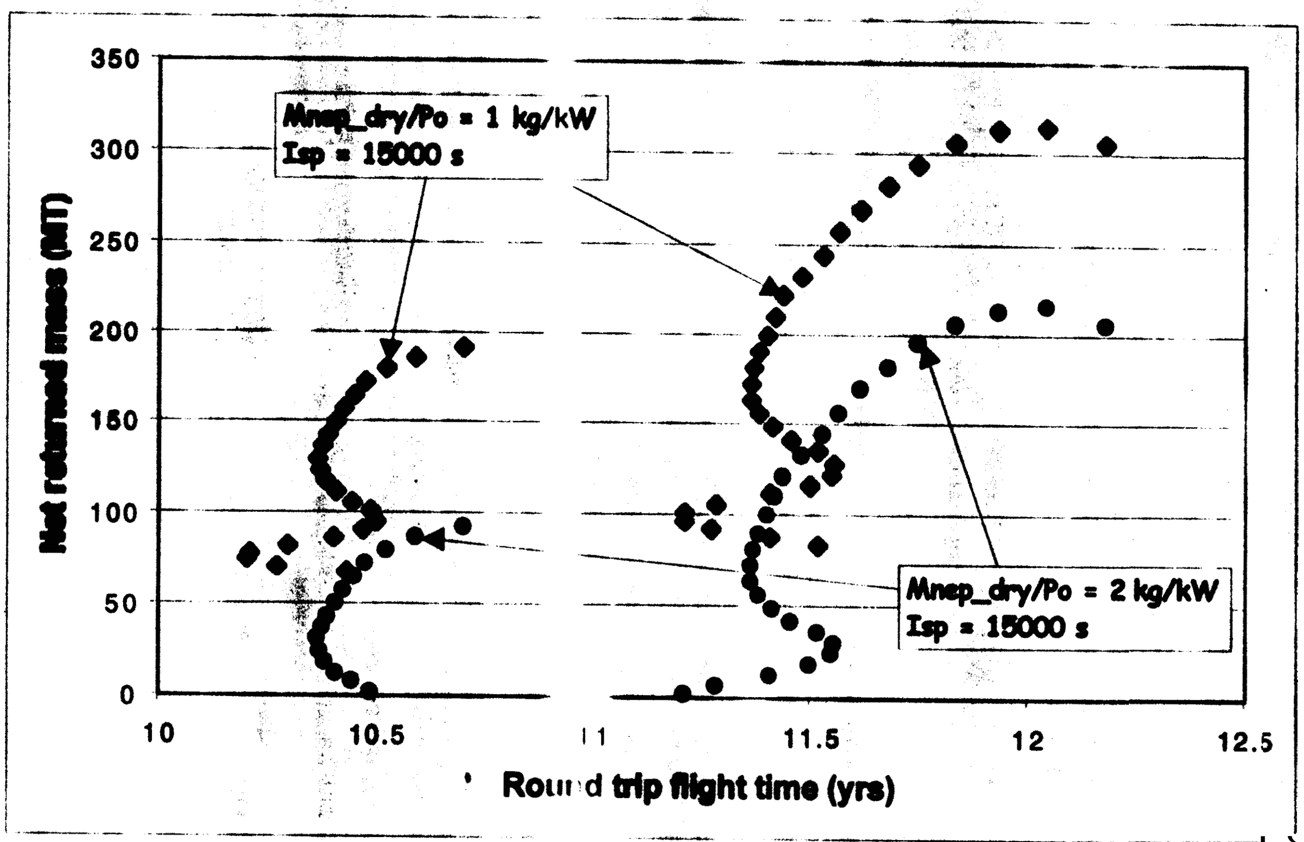
100-MWe: Mssion Results

Neptune Piloted Mission



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Conclusions

- NEP vehicle specific mass:

Power Level (MWe)	0 1	1	10	20	50	100
M(nep-tk)/Po (kg/kWe)	32	7.5	3.8	2.7	1.8	1.0

- NEP capabilities are promising
  - 9-12 years Pluto RV, 10-13 years Saturn/Titan SR, 3-4 years Europa mission, 5-6 years Europa SR...
- Delta IV launch
- Maximum Isp of 16000 s penalized high power missions (> 1 MWe)
  - Preliminary assessment of the technology (ion engines)
  - Set new requirements for the technology -> Isp ~ 30000-40000 s
- MPDT technology needs high effort to reduce Power Processing and especially Cabling masses